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THEORETICAL SUBSONIC PERFORMANCE OF A DUCTED ROCKET

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Date: JUL 1 1946 By: 95 Mullen (MM)

H. J. Stewart
H. J. Stewart, Chief
Section 1

F. J. Malina
F. J. Malina, Acting Director,
Jet Propulsion Laboratory, GALCIT

F. J. Malina for
C. B. Millikan, Acting Chairman,
JPL Executive Board

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61 Pages

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I. SUMMARY

This report is concerned with an endeavor to reduce to calculation the optimum performance of a jet propulsion system consisting of a conventional rocket motor inserted in a ramjet-type duct - a so-called ducted rocket - when in subsonic flight at sea level.

Observations of liquid propellant rocket motors reveal that the exhaust gases burn for some distance downstream of the exhaust nozzle, and it is anticipated that a similar secondary combustion will take place in the atmospheric air flowing through the duct.

The restriction on the flight speed of the jet-duct combination allows the problem of thrust computations to be conveniently separated into three parts:

(A) The air flow through the diffuser

(B) The mixing and burning processes inside the combustion chamber

(C) The discharge of the working fluid through a suitable nozzle into the atmosphere.

The final results of the performance calculations are presented as curves giving the "augmentation factor" (the ratio of thrusts for the rocket-ramjet combination and the rocket motor alone) as a function of different independent variables.

Of prime interest is the static thrust of the ducted-rocket power plant. The preliminary performance study of this special case (which, in an expanded form, will be the subject of a later report) indicates that the propellant gases will induce and accelerate a flow of air through the duct, in this

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manner producing a thrust. A preparatory approximate calculation shows that under static conditions the thrust of a ducted rocket, without a nozzle, will be between 60 per cent and 75 per cent of that of the normal rocket. Injecting and burning additional gasoline in the combustion chamber will cause the static thrust of the ducted rocket to increase asymptotically to the full rated performance of the rocket motor alone.

The performance of the rocket-ramjet combination in flight is encouraging. Among the examples chosen to illustrate applications of the general performance equations is that of a ducted rocket wherein the total energy of the rocket exhaust gases is released as a consequence of their combustion with the atmospheric oxygen flowing through the duct. Thrust computations for this case reveal that a suitable ramjet-type duct surrounding a liquid-propellant rocket may be instrumental in causing the thrust of the normal rocket to be multiplied almost fourfold. Compared to a ramjet of equal diffuser area ratio, equal combustion chamber area, and equal consumption of fuel, the ducted rocket has a greater thrust over the whole subsonic range of flight velocities.

II. INTRODUCTION

The ramjet, or athodyd (aero-thermodynamic-duct), consists of a diffuser in which atmospheric air is compressed, a combustion chamber within which a suitable fuel is injected into the air and burned, and a nozzle through which the products of the subsequent reaction are expelled. The simplest thermojet machine, after the rocket, is the ramjet, since it requires no device for drawing air through the duct entrance, the air being forced into the diffuser by the ramming action resulting from the forward motion of the unit.

On the basis of a theoretical performance analysis, it is anticipated that a large field of application to missiles and transonic and supersonic aircraft awaits the development of the ramjet power plant (Cf Refs 1 and 2).

Nevertheless, since the compression resulting from the ramming of air is the chief factor in preventing the products of combustion from discharging through the forward (diffuser) opening, forcing them to be exhausted by way of the nozzle, it is apparent that at zero airspeed the ramjet with an open diffuser entrance can produce no useful forward thrust.

Hence it is necessary to solve the problem of accelerating a ramjet off the ground and increasing the velocity to that at which combustion will proceed satisfactorily and a net forward thrust will be maintained. The take-off and initial stage of a ramjet-powered flight could be executed in an obvious way by the use of an auxiliary power plant, such as externally fitted rocket boosters, as in the now familiar JATO installations.

However, H. S. Tsien of the Jet Propulsion Laboratory, GALCIT, suggested that a rocket motor discharging into the forward part of the ramjet combustion chamber could not only supply the thrust required to accelerate the ramjet up to a velocity at which compression by ram would become effective, but also might act as an ejector jet, inducing a flow of air which would perhaps even augment the thrust of the normal rocket.

Such a combination of a conventional rocket motor and a ramjet duct (consisting of diffuser, combustion chamber, and discharge nozzle) will be termed a <u>ducted rocket</u>. The unit could be designed to jettison the rocket automatically when the ramjet thrust reached effective proportions, or the rocket motor could be built as an integral part of the duct structure.

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The elementary theory of the thrust augmentor is based on the following identities, which relate the kinetic energy of a mass m moving with a velocity v to the momentum of the same mass:

$$\frac{1}{2}mv^2 = (mv)\frac{v}{2} = \frac{(mv)^2}{2m}.$$

Therefore, corresponding to a given kinetic energy, the momentum is relatively great when the "vis viva" is diffused through a large mass of fluid moving comparatively slowly, and relatively small when the energy is concentrated in a slight mass possessing a high velocity.

The ordinary augmentor, which is based on the principle above, seeks to enhance the thrust of a given jet by diffusing the kinetic energy of the exhaust into a larger mass of fluid, the work required for the acceleration of the secondary flow being drawn from the otherwise wasted kinetic energy of the jet, which would trail away behind without contributing to the thrust.

For the conventional type of thrust augmentor, analytical and experimental studies indicate that only the thrust of a static jet, "when the induced air stream reaches the ejector at about atmospheric pressure and with negligible velocity", can be considerably augmented (Cf Ref 3). Further, "when the approach pressure or velocity of the air increases, the augmentation of thrust falls off rapidly; and it appears from these tests that, with a jet velocity of 1000 ft per sec, an approach velocity of 100 mph is sufficient to reduce the gain to a negligible amount. At higher approach velocities the effect of the ejector is practically nil, there being no appreciable gain or loss" (Cf Ref 3).

The results quoted above were obtained with augmentors employing converging guides ahead of the mixing section; the ducted rocket, which has a

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diverging section in front of the mixing chamber, thus differs from the augmenter in an important respect. Hence, it cannot be anticipated that at low speeds the ducted rocket thrust will be superior to that of the free jet; in fact, at small flight velocities the influence of the duct surrounding the rocket is such as to cause the thrust of the latter to decrease. The amount by which the thrust of the normal rocket is reduced at low speeds is an important problem which will be dealt with in a subsequent report.

Besides offering a solution to the problem of launching a ramjet-powered aircraft, a rocket motor permanently installed within a duct possesses certain attractive features of its own. Not the least of these is the possibility of using the flaming gases which issue from liquid propellant rockets as the means of starting combustion and of maintaining continuous ramjet operation.

Theoretical and experimental studies of rocket motors utilizing liquid fuels reveal that the exit gases contain large quantities of hydrogen and carbon monoxide (in addition to incombustibiles such as nitrogen, water vapor, and carbon dioxide [Cf Ref 4]). The heats of combustion of these gases are high; for example, the exhaust of a rocket motor designed for a mixture of liquid oxygen-gasoline, in the ratio 3:2 by weight, at a chamber pressure of 300 psia, yields 5600 Btu per lb of primary reactants when completely burned. The long jet of flame observed issuing from liquid propellant rockets is the result of the reaction of the H_2 and CO with oxygen in the air.

In practice this energy is wasted, so a device (such as a duct surrounding the jet) which could develop added thrust efficiently from the heat released by the secondary combustion of the propellant gases with air would be an improvement over existing rocket installations, if the increased drag due

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to the duct did not render the scheme impractical.

Moreover, as subsequent calculations will show, at higher flight speeds more air enters the ramjet combustion chamber than is required for the complete (secondary) burning of the rocket exhaust gases. Then, to utilize the excess inflowing air to better advantage, it might prove desirable to inject additional fuel, such as gasoline, into the combustion chamber.

From the foregoing, it is clear that energy can be added to the air flowing through the duct in either one or both of two ways:

(1) By burning the combustible rocket exhaust gases in the air

(2) By burning additional fuel in the air.

Essentially, the two processes differ only in the respect that the jet gases in (1), entering the combustion chamber at a velocity greater than that at which the fuel in (2) would be injected, possess more momentum than the added fuel. But even the momentum of the exhaust gases in (1) is small compared to the sum of momenta and pressure forces acting on the primary and secondary fluids in the combustion chamber, the ratio of the two having the order of 1 to 20. Hence, in the first approximation both means of adding energy to the duct air are considered to be equivalent, and the exact manner in which the heat is introduced is not important.

In any event, a performance analysis of the ducted rocket seemed warranted, and this report presents the first results of such a study.

It is reasonable to assume (as the optimum condition under which only the rocket exhaust is burned) that the propellant gases are completely consumed in the presence of the atmospheric oxygen flowing through the duct and to compute the performance of the ducted rocket on this premise. For the most

part, the calculations to follow are predicated on this assumption.

The middle portion of the duct will be designated the combustion chamber, since it is primarily used as the section wherein the propellant gases are burned.

Further, the rocket exhaust gases will be called the primary fluid; the inflowing air will be referred to as the secondary fluid; and the mixture of unburned air and products of the secondary reaction will be called the working fluid.

There was no a priori indication that the ducted rocket could compete with other jet propulsion schemes, so all calculations were directed toward evaluating the upper limit to the performance of the rocket-ramjet duct combination. Hence, all thermodynamic processes are considered perfect, frictionless flow of ideal compressible fluids is assumed throughout, heat losses are neglected, and chemical reactions are postulated to be carried to completion.

All computations are made for subsonic free flight velocities at sea level altitude, the latter restriction being arbitrary and serving only to particularize the results.

III. NOTATION

The notation adhered to in this report is summarized below.

A Ratio of the mass of air allowed to enter the duct to the mass of air required for complete secondary combustion of the rocket exhaust gases.

c Effective exhaust velocity of the rocket jet, ft per sec

c_p Specific heat of gas at constant pressure, Btu per lb ^{o}R

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c_v Specific heat of gas at constant volume, Btu per lb ^oR

D Drag force

E Total energy of gas, per lb mass

f Area, sq ft

f_t Cross sectional area of throat of the rocket nozzle, sq ft

F Thrust, lb

g Acceleration of gravity, 32.2 ft per sec^2

H Heat added to the working fluid in the combustion chamber, over and above that present in the rocket exhaust, Btu

m Mass of fluid flowing past a given plane in unit time, slugs per cu ft

M Mach number: the ratio of the fluid velocity to the local velocity of sound

p Absolute pressure, lb per sq ft

r Ratio of the mass of air entering the combustion chamber to the mass of burnt rocket gases ejected through the rocket nozzle, all in unit time

R Gas constant, Btu per slug ^oR

T Absolute temperature, ^oR

v Velocity, ft per sec

γ Ratio of specific heats of a gas, c_p/c_v

η Efficiency factor

ϱ Density, slugs per cu ft.

The subscripts c, o, 1, 2, 3, 4 have the following significances:

$(\ )_c$ will denote conditions in the rocket combustion chamber

$(\ )_o$ will denote conditions in the standard atmosphere, far from the

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disturbing effect of the duct

$( )_1$ will denote conditions at the entrance to the diffuser (Cf Fig 1)

$( )_2$ will denote conditions at the end of the diffuser, assumed to be in the same plane as the exit of the rocket nozzle and the entrance to the combustion chamber

$( )_3$ will denote conditions at the conclusion of the combustion chamber, assumed to be also the entrance to the duct nozzle

$( )_4$ will denote conditions at the exit of the duct nozzle.

The superscripts $( )$, $( )'$, $( )''$ will signify the following:

$( )$ will denote conditions referring to air, the secondary fluid

$( )'$ will denote conditions referring to the gases in the rocket jet, the primary fluid

$( )''$ will denote conditions referring to the working fluid.

For example,

γ, equal to 1.4, is the (constant) ratio of specific heats for air

γ', equal to 1.285 (Cf Ref 3), is the assumed constant ratio of specific heats of the rocket exhaust gases

γ'', the ratio of specific heats for the working fluid, is assumed to be a mean value, given by

$$\gamma'' = \frac{m_2\gamma + m_2'\gamma'}{m_2 + m_2'}.$$

In conformity with the notation, m_2 and m_2' are the masses of air and propellant gases, respectively, entering the ramjet combustion chamber in unit time.

IV. OVER-ALL PERFORMANCE OF A DUCTED ROCKET

The idealized cross section of what might be a typical ducted-rocket power plant is shown in Fig 1.

If it is assumed that all of the compression takes place in the diffuser (merely to achieve simplicity of ideas and language), air with a velocity v_o created by the motion of the aircraft enters the ramjet duct at the pressure p_o of the atmosphere. The diffuser is considered to compress the air adiabatically, the final values of the velocity and pressure being v_2 and p_2 , respectively.

After exhausting into the combustion chamber, the rocket gases are burned in the presence of air aspirated through the duct entrance, the energy released going partly to elevate the temperature of the working fluid, and partly to accelerate that mixture.

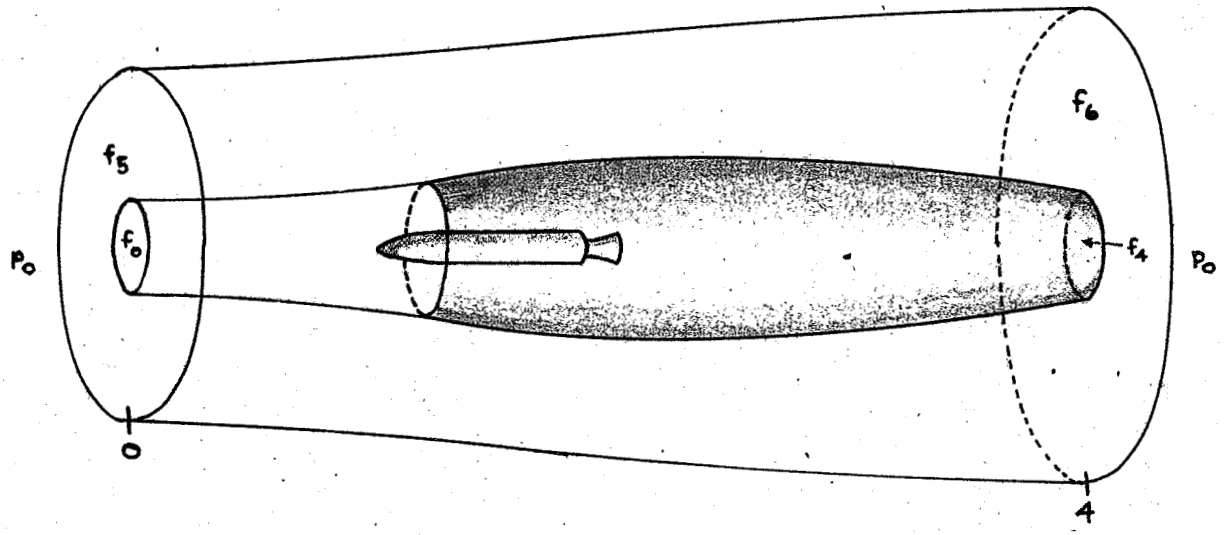
Because of the dissipation of energy in turbulent mixing, friction, and the acceleration of the working fluid, there will be a certain drop in the pressure of the mixture, as it passes through the combustion chamber, from the value p_2 to p_3'' .

The nozzle then expands the gases from the pressure p_3'' at the end of the combustion chamber to that of the atmosphere, p_o . After leaving the duct, the exhaust gases cool down to the temperature of the atmosphere, wasting their heat energy. The thrust augmentor functions to transfer efficiently as much as possible of the momentum and energy of the jet to a greater mass of air, in an endeavor to increase the momentum. Since the mass of matter leaving the jet is small compared with the mass of air flowing through the

duct, the increase in momentum is primarily dependent upon the amount by which the velocity at the ramjet nozzle exit exceeds the flight velocity of the duct. Under suitable conditions the exhaust velocity v_4'' will be greater than the free flight speed v_0; a thrust, or force in the direction of motion of the duct, is the direct result of the positive change in momentum, as will be shown below.

Rather than calculate the thrust of a ducted rocket (or of any reaction propulsive system) as the amount by which the momentum of the working fluid leaving the nozzle is greater than the momentum of air entering the diffuser, it is more accurate to compute the thrust by a consideration of the forces acting on the fluid passing through and over the duct.

In the sketch below the stream tube is taken so far in front of and away from the duct that the disturbing effect of the latter on the pressure has vanished. There is circular symmetry at every plane section normal to the axis, and outside of the tube free stream conditions obtain,



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The net thrust of the ducted rocket - the actual thrust less the drag - is then the excess of pressure force and momentum of the fluid at Section 4, which contains the nozzle exit, over the pressure forces and axial momenta of the fluid at the lateral stream tube surface and Section 0, where the outside air diffusion commences.

Since the quantity of mass leaving the duct is greater than that which enters, by an amount equal to the mass of propellant gas ejected from the rocket, the sum of areas $f_4 + f_6$ will be greater than the total area $f_0 + f_5$.

By definition, no fluid passes through the lateral surface of the tube, so that no momentum is transferred normal to the wall. However, because the generators of the tube surface are not parallel to the axis, there will be a component of pressure acting in the direction parallel to the axis. Further, since the pressures at Sections 0 and 4 are the same, p_o, the pressure force on Section 4 exceeds that on Section 0 by an amount given by the product of p_o and the difference in the areas at the two sections.

The magnitude of the difference in areas at Sections 0 and 4 is precisely the projection of the lateral surface of the stream tube on Section 4, and therefore the pressure force over the area by which $f_4 + f_6$ exceeds $f_0 + f_5$ is equal and opposite to the pressure force on the stream tube's lateral surface.

Upon application of the general thrust formula, which states that the net thrust is the difference between the pressure force and momentum at Section 4 and the axial components (denoted by a second subscript α) of the pressure forces and momenta taken over the lateral stream tube surface (f_L) and Section 0, it is clear that the

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$$\text{Net thrust} = \int_{f_4}(p_4+\rho_4 v_4^2)\,df + \int_{f_6}(p_6+\rho_6 v_6^2)\,df - \int_{f_0}(p_0+\rho_0 v_0^2)\,df - \int_{f_5}(p_5+\rho_5 v_5^2)\,df - \int_{f_L}(p_{La}+\rho_L v_L v_{La})\,df,$$ *

a positive thrust being a force urging the unit in a direction opposite to that of the main flow.

Since the integral of a sum is the sum of integrals, the ralation above may be rewritten as

$$\text{Net thrust} = \int_{f_4} p_4\,df + \int_{f_4}\rho_4 v_4^2\,df + \int_{f_6} p_6\,df + \int_{f_6}\rho_6 v_6^2\,df - \int_{f_0} p_0\,df - \int_{f_0}\rho_0 v_0^2\,df - \int_{f_5} p_5\,df - \int_{f_5}\rho_5 v_5^2\,df - \int_{f_L} p_{La}\,df - \int_{f_L}\rho_L v_L v_{La}\,df.$$

The pressures over the areas f_0, f_4, f_5, f_6 are the same and equal to p_0, so

$$\int_{f_4} p_4\,df + \int_{f_6} p_6\,df - \int_{f_0} p_0\,df - \int_{f_5} p_5\,df = \int_{f_4+f_6-f_0-f_5} p_0\,df \quad ;$$

according to one of the considerations above, this force is equal and opposite to $\int_{f_L} p_{La}\,df$, the axial component of the pressure force upon the lateral surface of the free stream tube, and hence there remains

$$\text{Net thrust} = \int_{f_4}\rho_4 v_4^2\,df + \int_{f_6}\rho_6 v_6^2\,df - \int_{f_0}\rho_0 v_0^2\,df - \int_{f_5}\rho_5 v_5^2\,df.$$

Here the condition $\int_{f_L}\rho_L v_L v_{La}\,df = 0$, which means that there is no axial momentum flux through the walls, has been imposed.

In the usual definition the thrust of a reaction propulsive system is given as the excess of momentum leaving the system over that which enters. With this interpretation the formula for the thrust of the ducted rocket consists of the terms

$$\int_{f_4}\rho_4 v_4^2\,df - \int_{f_0}\rho_0 v_0^2\,df,$$

---

*The axial component of momentum flux through the lateral surface of the stream tube is represented as $\int_{f_L}\rho_L v_L v_{La}\,df$, being the integral, over the lateral surface, of the mass flow through the surface multiplied by the axial component of the velocity.

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or the change in momentum of the fluid passing through the duct. The quantity $\int_{f_6} \rho_6 v_6^2 df - \int_{f_5} \rho_5 v_5^2 df$, representing the momentum change experienced by the fluid in passing over the duct, is essentially the external drag force (in the absence of a pressure difference between Sections 6 and 5) and will not be explicitly calculated, since the magnitude will depend upon the size and configuration of a particular installation and its relationship to the other component parts of a projectile or aircraft.

Then the force exerted on the fluid passing through the duct, heretofore called the thrust, is given by

$$F = \int_{f_4} \rho_4 v_4^2 df - \int_{f_0} \rho_0 v_0^2 df.$$

It should be noted that this quantity is only defined as the thrust and does not represent the net force imparted to the duct by the air affected as a result of the motion of the duct through the atmosphere.

V. PERFORMANCE CALCULATIONS

The theoretical performance analysis of the ducted rocket is conveniently divided into three parts:

(A) Properties of the air entering the combustion chamber are obtained in terms of free stream conditions.

(B) Characteristics of the working fluid leaving the combustion chamber are calculated as functions of properties of the primary and secondary fluids entering the middle section.

(C) The nozzle is designed to reduce the pressure of the working fluid to that of the atmosphere.

These separate parts of the investigation are reported upon below.

From the idealized cross section of the ducted rocket power plant, Fig 1, it is evident that the large end of the diffuser and the entrance to the combustion chamber are assumed to be one and the same section; a like statement is true of the combustion chamber exit and the nozzle entrance.

Since "the force exerted by a fluid on a body [depends] ... only on the relative velocity between them" (Cf Ref 5), it follows that the ducted rocket may be chosen at rest, with the air moving over and through the duct.

At an infinite distance from the duct, and at the same height above the surface of the earth - here considered as plane - the velocity of the air is assumed to be uniform and parallel to the axis of the rocket. Atmospheric conditions are taken as standard, based on the NACA Atmospheric Tables (T.R. No. 218, N.A.C.A., 1925); furthermore, only subsonic flight velocities are considered.

In all performance calculations sea level atmospheric properties will be used, for which

Acceleration of gravity = g = 32.2 ft per sec^2

Density = ρ_o = 0.002378 slugs per cu ft

Pressure = p_o = 2116 lb per sq ft

Ratio of specific heats for air = γ = 1.4

Temperature = T_o = 518.4 ^{o}Rankine

Velocity of sound = $\sqrt{\gamma \frac{p_o}{\rho_o}}$ = 1120 ft per sec.

A. <u>Calculation of the Properties of Air Entering the Combustion Chamber in Terms of Free Stream Conditions</u>

The calculation of the properties of air entering the combustion chamber

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should be separated into two parts: first, the free stream diffusion ought to be discussed, and second, properties of the air at the diffuser exit should be given as functions of those at the diffuser entrance.

To render possible the analysis of the free stream diffusion, given properties of air at the diffuser entrance in terms of free stream conditions, the following idealizations are employed:

(1) The air is postulated to be a perfect gas with constant specific heats

(2) All thermodynamic processes are considered to be adiabatic with the exponent 1.4

(3) The free stream diffusion is assumed 100 per cent efficient.

Further, only subsonic flight velocities of the ducted rocket are considered; hence no shocks can occur in front of the diffuser.

If the additional requirement is imposed that the physical processes are restricted to those which conserve mass, a set of simultaneous equations to be solved for M_1 and $\frac{T_1}{T_0}$ in terms of the parameters M_0 and $\frac{f_1}{f_0}$ are obtained.

However, the analysis of expanding fluid flow in a duct is more complex, because even a well designed diffuser will have inevitable pressure losses, the presence of which tend to make the transformation of kinetic energy into pressure energy less efficient than it would be where there is no degradation of energy. In this way the concept of diffuser efficiency, originally suggested by Stodola, is introduced, being defined as

$$\eta_D = -\frac{dp}{\rho d(v^2/2)},$$

representing the fraction of kinetic energy change useful for purposes of compression (referred to 1 mol of fluid).

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Because of the great variety of uses to which subsonic diffusers may be put, there is available a large amount of experimental data on the performance and efficiency of diverse shapes and types of devices for transforming part of the kinetic energy of a flowing fluid to pressure energy. The most complete coalation and examination of data are contained in Refs 6 and 7, which include also a number of references to the literature.

The diffuser efficiency depends primarily upon the rate at which the flow expands; for a conical diffuser with straight diverging walls, the optimum angle between the walls lies in the interval $6^{\circ} - 8^{\circ}$. Diffusers with non-circular cross sections are less efficient than corresponding flow expanders which have circles as their right sections.

Nevertheless, because it is so important to keep the weight and drag of the ducted rocket power plant as low as possible, it may not be practical to use a small angle of expansion, since that would require a diffuser of prohibitively great length; it may be necessary, therefore, to employ a diffuser with a large angle of divergence.

According to Ref 7, however, because of the free stream expansion a diffuser with an angle of divergence greater than the optimum may be used without the penalty of large losses, as the external diffusion acts to "assist the internal flow by fanning out the streamlines so that they tend to follow more easily the diverging walls of the diffuser. Good efficiency may therefore be expected from a diffuser which combines an external with an internal expansion".

Fig 14 of Ref 2, illustrating the variation in η_D (the diffuser efficiency) as a function of the Mach number M_1 of the entering fluid, summarizes the

available data on diffuser performance. Up to speeds approaching that of sound the diffuser efficiency is essentially constant, equal to 0.85.

Computations of the ducted rocket thrust, assuming diffuser efficiencies of 100 per cent and 85 per cent, differed only by at most 9 per cent. Further, in this report all calculations are intended to reveal the optimum performance of a ducted rocket, and therefore the mathematically maximum efficiency of the diffuser ought to be that used in the numerical work. Consequently, the diffuser efficiency will be taken as 100 per cent, the same as that of the free stream diffusion.

Hence there is no essential difference between the compression as realized in the free stream and as accomplished in the ramjet diffuser; so the diffusion can be considered to take place entirely in the atmosphere, resulting in formulae which give conditions at the combustion chamber entrance directly in terms of those prevailing in the free stream.

The idealizations listed at the beginning of this Part, now applied to the <u>total</u> expansion of the flow, together with Bernoulli's equation for the isentropic flow of a compressible fluid and the Law of Conservation of Mass, are expressed mathematically below:

$$\frac{T_2}{T_0} = \frac{1 + \frac{\gamma-1}{2} M_0^2}{1 + \frac{\gamma-1}{2} M_2^2} \qquad \text{(Bernoulli's equation)}$$

$$\frac{V_2}{V_0} = \frac{f_0}{f_2}\frac{\rho_0}{\rho_2} = \frac{1}{f_2/f_0}\frac{1}{\rho_2/\rho_0} \qquad \text{(Conservation of mass)}$$

$$\frac{\rho_2}{\rho_0} = \frac{p_2}{p_0}\frac{T_0}{T_2} = \left(\frac{p_2}{p_0}\right)^{\frac{1}{\gamma}} \qquad \text{[Assumption (1)]}$$

$$\frac{p_2}{p_0} = \left(\frac{T_2}{T_0}\right)^{\frac{\gamma}{\gamma-1}} \qquad \text{[Assumptions (2) and (3)]} .$$

These relations are easily combined to give a set of simultaneous equations which can be solved for M_2 and $\frac{T_2}{T_0}$ in terms of the parameters M_0 and $\frac{f_2}{f_0}$:

$$M_2 = \frac{\frac{1}{f_2/f_0}}{(T_2/T_0)^3} M_0$$

$$\frac{T_2}{T_0} = \frac{1+0.2M_0^2}{1+0.2M_2^2} ;$$

in these relations γ has been set equal to 1.4, following idealization (2). For given M_0 and $\frac{f_2}{f_0}$ the values of M_2 and $\frac{T_2}{T_0}$ can be easily calculated by an iteration process. Then the values of $\frac{p_2}{p_0}$ and $\frac{\rho_2}{\rho_0}$ can be computed; with this knowledge the results can be readily checked by calculating M_2 from the formula

$$M_2 = \frac{1}{f_2/f_0} \frac{1}{\rho_2/\rho_0} \frac{1}{\sqrt{T_2/T_0}} M_0 ,$$

derived simply from the Law of Conservation of Mass:

$$\rho_0 f_0 V_0 = \rho_2 f_2 V_2 .$$

B. Characteristics of the Working Fluid Leaving the Combustion Chamber in Terms of Individual Properties of the Entering Air and Rocket Exhaust Gases

1. The combustion chamber

The combustion chamber of the ramjet component of the ducted rocket is considered to be a circular cylinder, of volume sufficient to provide space for combustion, and of length which permits complete secondary burning of the rocket exhaust gases before the products of the secondary reaction enter the duct nozzle. In addition, the combustion chamber should be long enough to insure that the velocity distribution across the end section is reasonably uniform, a strictly constant velocity distribution being that for which the losses are a minimum.

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Such geometrical specialization is valid because the theoretical analysis of the ducted rocket is independent of any particular combustion chamber shape, except that a uniform cross section is assumed throughout, requiring only that the chamber be so designed that mixing and combustion are complete by the time the working fluid reaches the duct nozzle.

The large amount of momentum present in the rocket jet, by providing a source of high-energy turbulent eddies, should materially assist the mixing process.

Theory has not been successful in predicting the length of chamber necessary to secure complete combustion for given entrance conditions, propellant, and air flow. Also, because this problem is so new, there has not been sufficient time to reduce experimental data to design parameters and working rules. Hence no estimate will be given of the length of chamber required to meet the objectives of complete mixing and complete combustion. It may be possible to reduce the chamber length considerably by the installation of multiple rockets, which would function to assist the mixing and burning processes.

For example, using multiple fuel injectors, recent ramjet combustion chamber tests conducted at the Jet Propulsion Laboratory, GALCIT, seem to place the length of chamber necessary to secure reasonably complete mixing and burning at 40 in, independent of the diameter.

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It will be necessary to give careful consideration to the combustion chamber length, since the external drag is of outstanding importance; for this reason a chamber with a given cross sectional area should have the minimum possible length, with due regard to other factors such as weight, simplicity of construction, fuel consumption, etc. The velocity of the air flowing through the central duct section is small, and hence a judicious design will produce a smooth, low-drag, internal surface.

Reference 8 presents the theoretical performance analysis of a system employing a rocket motor in a duct, acting as an ejector, to provide the source of power for driving propellant pumps. Such a system is similar to that of the ducted rocket, and hence the reference above will serve as a guide in the performance analysis to follow, as the same basic equations (with one slight modification) govern the phenomena.

For the calculations relative to the processes inside the combustion chamber, the following idealizations are employed:

(a) The air, the rocket exhaust gases, and the working fluid are postulated to be perfect gases with constant specific heats.

(b) The mixing of the products of the secondary burning is considered complete at the end of the combustion chamber, where the distribution of velocity is uniform.

(c) Frictional losses are non-existent.

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(d) At Section 2 the pressure in the rocket jet is assumed to be that of the air in the same plane; the velocity of the jet at this section is defined to be the exhaust velocity. Since the pressure at the combustion chamber entrance will increase with the free flight velocity of the unit, it follows that the pressure in the jet will vary, causing the area of the reference section (f_2') to change with v_o. However, since f_2' appears always as part of the expression $f_2 + f_2'$, where f_2 is at least $200\, f_2'$, it follows that variations in the size of f_2' will not be significant. Hence in the numerical work f_2' will be chosen as a constant, intermediate between the values it assumes for $v_o = 0$ ($M_o = 0$) and $v_o = 1120$ ft per sec ($M_o = 1$).

2. Mathematical formulation of the problem

The fundamental relations governing the mixing process are as follows:

(a) Condition of conservation of mass

The mass of air and propellant gases entering the combustion chamber in unit time must be equal to the mass of reaction products passing through the end section in unit time.

The requirement that the mass be conserved is imposed by setting

$$\rho_2 f_2 v_2 + \rho_2' f_2' v_2' = \rho_3'' f_3'' v_3'',$$

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or

$$m_2 + m_2' = \rho_3'' f_3'' v_3''. \qquad (1)$$

(b) Condition of conservation of pressure forces and momentum

The difference between the momenta of the air and propellant gases entering the combustion chamber, and the momentum of the working fluid passing through the end section, all in unit time, must equal the sum of the forces acting on the flow, which are only the pressure forces over the cross section of the fluid, since the combustion chamber is considered to be a smooth cylinder.

The requirement that the sum of the momentum and pressure forces be conserved is imposed by setting

$$\rho_2 f_2 v_2^2 + \rho_2' f_2' v_2'^2 - \rho_3'' f_3'' v_3''^2 = p_3'' f_3'' - p_2 f_2 - p_2' f_2'.$$

Since $f_2 + f_2' = f_3''$ and $p_2' = p_2$,

$$m_2 v_2 + m_2' v_2' - (m_2 + m_2') v_3'' = f_3''(p_3'' - p_2) \qquad (2)$$

(c) Condition of conservation of energy

The total energy of the air and rocket gases entering the combustion chamber, plus the heat, H, released during the secondary burning of the rocket exhaust or additional fuel in the inflowing air,

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must equal the total energy of the reaction products leaving the combustion chamber. Because of the idealization (a) of the previous section (Cf p. 21), the enthalpy $c_p T$ for each fluid (air, propellant gas, working fluid) may be written as $\frac{\gamma}{\gamma-1}\frac{p}{\rho}$.

The requirement that the total energy be conserved is imposed by setting

$$m_2\left(\frac{\gamma}{\gamma-1}\frac{p_2}{\rho_2}+\frac{1}{2}V_2^2\right)+m_2'\left(\frac{\gamma'}{\gamma'-1}\frac{p_2}{\rho_2'}+\frac{1}{2}V_2'^2\right)+H=(m+m_2')\left(\frac{\gamma''}{\gamma''-1}\frac{p_3''}{\rho_3''}+\frac{1}{2}V_3''^2\right),$$

or

$$m_2E_2+m_2'E_2'+H=(m_2+m_2')\left(\frac{\gamma''}{\gamma''-1}\frac{p_3''}{\rho_3''}+\frac{1}{2}V_3''^2\right). \quad (3)$$

Summarizing, the basic relations controlling the mixing and burning processes in the combustion chamber are

$$m_2+m_2'=\rho_3''f_3''V_3''$$

$$m_2V_2+m_2'V_2'=(m_2+m_2')V_3''+(p_3''-p_2)f_3''$$

$$m_2E_2+m_2'E_2'+H=(m_2+m_2')\left(\frac{\gamma''}{\gamma''-1}\frac{p_3''}{\rho_3''}+\frac{1}{2}V_3''^2\right)$$

From Part A, properties γ, ρ_2, V_2, p_2, $m_2\,(=\rho_2 f_2 V_2)$ of the air entering the combustion chamber are known, and the section following will present formulae showing the relationship between γ', ρ_2', V_2', m_2' and the defining characteristics of a rocket motor.

Further, a given rocket motor will have a definite quantity of heat energy available in its

exhaust, as subsequent calculations will indicate, and so H is a known, or easily computed, number.

Also, $f_3'' = f_2 + f_2'$, where f_2' is a constant, somewhere intermediate between f_2' for $M_0 = 0$ and $M_0 = 1$; f_2 is, within practical limitations, a wholly arbitrary fundamental design variable.

Finally, as previously remarked, γ'' is taken as a mean value, given by

$$\gamma'' = \frac{m_2\gamma + m_2'\gamma'}{m_2 + m_2'}.$$

There remain only three unknowns, v_3'' , ρ_3'' , and p_3'' , which may be obtained as the solution of the set of three simultaneous equations (1), (2), (3). The value of v_3'' is given as

$$v_3'' = \frac{\gamma''}{(m_2+m_2')(1+\gamma'')}\left[p_2 f_3'' + m_2 v_2 + m_2' v_2' \pm \sqrt{(p_2 f_3'' + m_2 v_2 + m_2' v_2')^2 - 2(m_2+m_2')\frac{\gamma''^2-1}{\gamma''^2}(H + m_2 E_2 + m_2' E_2')}\right] \quad (4)$$

where

$$E_2 = \frac{\gamma}{\gamma-1}\frac{p_2}{\rho_2} + \frac{1}{2}v_2^2$$

$$E_2' = \frac{\gamma'}{\gamma'-1}\frac{p_2}{\rho_2'} + \frac{1}{2}v_2'^2.$$

The significance of the (±) sign will be discussed later. With the value of v_3'' given above, the magnitudes of ρ_3'' and p_3'' are computed using (1) and (2) respectively:

$$\rho_3'' = \frac{m_2 + m_2'}{f_3'' v_3''} \quad (5)$$

$$p_3'' = \frac{p_2 f_3'' + m_2 v_2 + m_2' v_2'}{f_3''} - \frac{m_2 + m_2'}{f_3''} v_3'' \quad (6)$$

The temperature T_3'' of the reaction products at the end of the combustion chamber may be computed from

$$T_3'' = \frac{p_3''}{R_3'' \rho_3''}, \qquad (7)$$

following from the assumption that those reaction products obey the equation of state for perfect gases. Here R_3'', the equivalent gas constant for the working fluid, can be obtained from

$$R_3'' = \frac{1}{m_2 + m_2'} \sum_i m_i'' R_i'',$$

in which m_i'' and R_i'' are respectively the masses and individual gas constants of the constituents of the reaction products.

Now, the addition of heat to a gas flowing initially at subsonic speed in a straight pipe serves to increase the temperature, pressure, and velocity; the speed may jump to a supersonic value with the instantaneous addition of a sufficient quantity of heat. But if the heat is added in a continuous fashion the velocity discontinuity cannot occur, and the speed will never become supersonic, the maximum attainable velocity being that of sound. Since the fluids entering the combustion chamber have subsonic velocities and a combustion process generally evolves

heat in a continuous manner, it is impossible for the working fluid to attain a supersonic speed at the combustion chamber exit, although a sonic velocity is not excluded.

Formula (4) contains a (±) sign because it is the solution of a quadratic equation, but only the smaller root, using the (-) sign, is of practical significance. This is true because when the (-) sign is used in (4) the Mach number $M_3'' = \frac{V_3''}{\sqrt{\gamma'' P_3''/\rho_3''}}$ will always be less than 1, while the (+) sign invariably causes M_3'' to be greater than unity.

Therefore the larger root in formula (4) - corresponding to the (+) sign -, which results in the unattainable supersonic velocity, must be discarded as physically not significant (for the ducted rocket with secondary burning proceeding in an orderly way).

Finally, the conditions at the end of the combustion chamber are given as single-valued functions of properties of the entering fluids:

$$V_3'' = \frac{\gamma''\left[P_2 f_3'' + m_2 v_2 + m_2' v_2' - \sqrt{(P_2 f_3'' + m_2 v_2 + m_2' v_2')^2 - 2\frac{\gamma''^2-1}{\gamma''^2}(m_2+m_2')(H + m_2 E_2 + m_2' E_2')}\right]}{(1+\gamma'')(m_2+m_2')} \qquad (8)$$

$$\rho_3'' = \frac{m_2+m_2'}{f_3'' V_3''} \qquad (9)$$

$$P_3'' = \frac{P_2 f_3'' + m_2 v_2 + m_2' v_2'}{f_3''} - \frac{m_2+m_2'}{f_3''} V_3'' \qquad (10)$$

It should be recalled that

$$E_2 = \frac{\gamma}{\gamma-1}\frac{p_2}{\rho_2} + \frac{1}{2}v_2^2, \quad E_2' = \frac{\gamma'}{\gamma'-1}\frac{p_2}{\rho_2'} + \frac{1}{2}v_2'^2.$$

3. The rocket motor

So far no numerical values have been set into the expressions (8), (9), and (10) above.

However, from the diffuser calculations, Part A, the magnitudes of γ, v_2, p_2, ρ_2 are known.

The primed quantities, and H, refer to the rocket jet and can be computed as soon as the rocket motor is defined by the ratio of specific heats (γ') of its exhaust, propellants, thrust (F), chamber pressure (p_c), and effective exhaust velocity (c), equal to the actual exhaust velocity.

The rocket motor will now be specified as one which consumes a mixture of liquid oxygen and gasoline in the weight ratio of 3:2 at a chamber pressure of 300 psia. For such a motor the effective exhaust velocity of the jet is 6440 ft per sec when the pressure of the air into which it discharges is 14.7 psia, and the ratio of specific heats (γ') of the propellant gas is 1.285. All of these data are from Table II of Ref 3. The rocket propellant, mixture ratio, and pressure are chosen because of the large amount of energy that can be released in the secondary burning.

Although the thrust of the rocket motor will change with the outside pressure, such variation will be slow, and the thrust will be assumed constant. The rocket will be considered to have a rated thrust of 500 lb, although the final result of the computations will present

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the ducted rocket thrusts in non-dimensional form as

$$\frac{\text{Thrust of ducted rocket}}{\text{Thrust of rocket}} = \text{Augmentation factor}$$

$$= \frac{\text{Effective exhaust velocity of ducted rocket}}{6440 \text{ ft per sec}}.$$

From the well-known theory of the de Laval nozzle, the performance of the rocket motor can be calculated from the formulae

$$F = m_2' v_2' \tag{11}$$

$$v_2' = \sqrt{\gamma' \frac{p_c}{\rho_c}} \sqrt{\frac{2}{\gamma'-1}\left[1-(p_2/p_c)^{\frac{\gamma'-1}{\gamma'}}\right]} \tag{12}$$

$$\frac{f_2'}{f_t} = \frac{\left(\frac{2}{1+\gamma'}\right)^{\frac{\gamma'+1}{2(\gamma'-1)}}}{(p_2/p_c)^{1/\gamma'}\sqrt{\frac{2}{\gamma'-1}\left[1-(p_2/p_c)^{(\gamma'-1)/\gamma'}\right]}} \tag{13}$$

$$m_2' = f_t\left(\frac{2}{\gamma'+1}\right)^{\frac{\gamma'+1}{2(\gamma'-1)}}\sqrt{\gamma' p_c \rho_c} \tag{14}$$

Here f_2' is the area of the right section of the jet where the ambient pressure is p_2.

In formula (12) $\gamma' = 1.285$ and $p_c = 300$ psia for all values of p_2 and v_2'. Since v_2' will change slowly with p_2 (for instance, when p_2=14.7 psia, v_2' =6440 ft per sec, while when p_2 = 27.2 psia [the largest value it assumes in the examples considered in this report], v_2' =5990) it appears reasonable to consider v_2' to be constant, equal to, say, 6440 ft per sec. With these values it is easy to compute ρ_c from (12). Since the flow aft of the rocket nozzle throat is supersonic, ρ_c will not be affected by changes in the pressure p_2, and hence will be constant.

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The area of the nozzle throat, f_t, which is necessarily fixed for a given rocket motor, can be computed from Eq (14) for a condition at which all of the other quantities in that relation are known; for example, when $p_2 = 14.7$ psia, $v_2' = 6440$ ft per sec, $m_2' = \frac{500}{6440}$ slugs per sec, $\gamma' = 1.285$, $p_c = 300$ psi, and $\rho_c = 0.00448$ slugs per cu ft [from Eq (12)], Eq (14) gives $f_t = 0.003359$ sq ft.

Further, formula (13) allows the calculation of $\frac{f_2'}{f_t}$ as a function of the outside pressure p_2, so that f_2' can be computed with the aid of the identity

$$f_2' = (f_2'/f_t) f_t .$$

Finally, ρ_2', the density of the rocket exhaust gases where the pressure is p_2 and the velocity is v_2', is given by

$$\rho_2' = \frac{m_2'}{f_2' v_2'}$$

by the Law of Conservation of Mass.

In this manner numerical sizes of all of the primed variables appearing in the basic formulae (8), (9), and (10) are determined as functions of the parameters M_o and $\frac{f_2}{f_o}$ and the free flight conditions p_o and ρ_o .

4. Numerical value of H, the heat added by completely burning the rocket exhaust in air

The heat energy that can be released by burning the rocket exhaust gases in the combustion chamber will naturally depend upon the products of the reaction between the liquid oxygen and gasoline in the rocket motor. The composition of those reaction products is given in Table I, Ref 3, part of which is repeated below.

TABLE I

Products of Combustion of Gasoline and Liquid Oxygen

Initial Temperature of Fuel: 350°F

Initial Temperature of Oxidant: -200°F

Mixture Ratio: Weight ratio, oxidant to fuel: 1.5:1

Reaction Pressure	Oxidant	Mixture Ratio	Composition - Mole Fraction			
			CO_2	CO	H_2O	H_2
300 psia	100% O_2	1.5	0.033	0.455	0.124	0.388

As the CO_2 and H_2O cannot be further oxidized, additional energy from the exhaust gases can only be obtained from the reaction between the CO and atmospheric oxygen and the H_2 and atmospheric oxygen. The particular combination of rocket propellant, mixture ratio, and chamber pressure employed in the calculations was chosen because it was the one with the largest percentage of CO and H_2 in the exhaust gases.

The heat energy in the exhaust is computed by means of the obvious formula

$$H = \sum_i w_i h_i ,$$

where H is the heat energy released in the secondary burning of the propellant gases, w_i and h_i being respectively the weights and heats of combustion per lb of the component combustible substances in the exhaust gases.

Each second the rocket motor ejects $\frac{500}{6440} \times 32.2 = 2.5$ lb of matter through its nozzle.

Then the weight of CO expelled from the rocket is 1.85 lb per sec, while the rocket ejects 0.1130 lb of H_2 every second; these numbers are calculated in the manner shown below. (The first number is the fractional part of a mole occupied by the substance mentioned on the same line, the second number is the molecular weight, and X is the factor which converts the product [mole fraction] • [molecular weight] to pound weight.)

Weight of CO_2 expelled from rocket = (0.033) (44) X = 1.45 x

Weight of CO expelled from rocket = (0.455) (28) X = 12.73 x

Weight of H_2O expelled from rocket = (0.124) (18) X = 2.23 x

Weight of H_2 expelled from rocket = (0.388) (2) X = 0.776 x

17.186 x

Since 17.19 X = 2.5, it follows that X = 0.1454. From this the numerical values given above follow immediately.

According to Ref 9, CO evolves 4.375 Btu per lb when completely burned to CO_2, while the combustion of H_2 releases 52,050 Btu per lb. Consequently, the heat energy available in the rocket exhaust is

$$H = (1.85)(4.375) + (0.1130)(52,050)$$

$$= 14,000 \text{ Btu per sec}$$

$$= 5600 \text{ Btu per lb of primary reactants.}$$

The comparative order of magnitude of this amount of energy will attain its proper significance if it is noted that the total energy in the rocket exhaust, before entering into the secondary

reaction with the air in the duct, is approximately 1700 Btu per lb of fuel. One lb of gasoline will yield approximately 20,000 Btu when it is completely burned.

Hence, even if a fraction of the latent heat energy in the exhaust is released by the secondary burning, the ducted rocket power plant should produce a thrust substantially greater than that of a normal rocket unit.

C. Calculation of Properties of the Working Fluid at the End of the Ramjet Nozzle (Section 4) in Terms of those at the Nozzle Entrance (Section 3)

In this Part the only unknown for which a value is sought is the velocity v_4'' at the exit of a nozzle designed to expand the pressure of the working fluid to that of the atmosphere. A given nozzle will properly expand the products of combustion for only one set of operating conditions.

Even so, the thrust calculations are based on the assumption that a flexible nozzle, which instantly adjusts itself to any given situation, is employed aft of the combustion chamber, although it is realized that such a device would unduly complicate an actual ducted rocket installation. Of course, this idealization has its origin in the endeavor to calculate the optimum thrust of the rocket-ramjet duct combination.

Analogous to the diffuser efficiency, η_D, defined in Part A, there is a so-called nozzle efficiency factor, η_N, described as the fraction of pressure change available for accelerating the fluid; thus,

$$\eta_N = -\frac{\rho\, d(v^2/2)}{dp}.$$

Experimental data show that a carefully designed discharge nozzle of a rocket motor will be free of shock waves and eddying losses and will have

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an efficiency of approximately 98 per cent. For the most part the flow in the duct nozzle will be subsonic, although the possibility of an acoustic velocity at the combustion chamber exit may arise. For the latter case the efficiency will be about 98 per cent; a subsonic nozzle does not even admit the possibility of shock waves, and so the efficiency may be even higher. These considerations justify assumption (4) below.

The flow through the discharge nozzle can be calculated in an elementary manner by the employment of the Law of Conservation of Mass and the following idealizations:

(1) The working fluid is postulated to be a perfect gas with constant specific heats.

(2) All thermodynamic processes are considered adiabatic. The exponent γ'', which applies here, can be taken as a mean value between γ and γ', with due regard to the mass of each component entering into the secondary burning; that is,

$$\gamma'' = \frac{m_2\gamma + m_2'\gamma'}{m_2 + m_2'}.$$

However, to simplify computations, γ'' will be assumed constant, equal to 1.4.

(3) The nozzle expands the working fluid from the pressure p_3'' at its entrance to the atmospheric pressure p_o at its exit without loss; η_N = 100 per cent.

The mathematical formulation of requirements (1), (2), (3) above,

$$\frac{T_4''}{T_3''} = \left(\frac{p_o}{p_3''}\right)^{\frac{\gamma''-1}{\gamma''}},$$

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when combined with a form of Bernoulli's equation for the isentropic flow of a compressible fluid,

$$\frac{T_4''}{T_3''} = 1 + \frac{\gamma''-1}{2} M_3''^2 \left[1 - (V_4''/V_3'')^2\right],$$

equivalent to that used in Part A, results in a single formula giving V_4'' in terms of properties of the fluid entering the nozzle:

$$V_4'' = V_3'' \sqrt{1 + \frac{1-(P_0/P_3'')^{\frac{\gamma''-1}{\gamma''}}}{\frac{\gamma''-1}{2} M_3''^2}}. \qquad (15)$$

The nozzle dimensions may be easily computed from the Law of Conservation of Mass:

$$f_4'' = \frac{f_3'' \rho_3'' V_3''}{\rho_4'' V_4''} = \frac{f_3''}{\frac{V_4''}{V_3''}\frac{\rho_4''}{\rho_3''}}.$$

Here f_3'' $(= f_2 + f_2')$ is known, V_4''/V_3'' is given from the previous result, and $\rho_4''/\rho_3'' = (P_0/P_3'')^{1/\gamma''}$ from the perfect gas and adiabatic flow conditions.

To recapitulate,

$$f_4'' = f_3'' \left(\frac{P_0}{P_3''}\right)^{-\frac{1}{\gamma''}} \left[1 + \frac{1-(P_0/P_3'')^{\frac{\gamma''-1}{\gamma''}}}{\frac{1}{2}(\gamma''-1) M_3''^2}\right]^{-\frac{1}{2}}.$$

VI. THRUST CALCULATIONS

With the value of V_4'' calculated from Eq (15), the thrust of the ducted rocket can be computed from the formula

Thrust = change in momentum of the fluid passing through the duct.

The momentum of the working fluid leaving the duct is $(m_2+m_1')v_4''$, while the momentum of air entering the ramjet is $m_2 v_0$ (if it is recalled that $m_0 = m_2$). Hence

$$\text{Thrust} = (m_2+m_1')v_4'' - m_2 v_0.$$

The computed thrusts are presented in the form of curves showing the augmentation factor as a function of different independent variables, the augmentation factor being the ratio of thrusts for the ducted and the unenclosed jet.

The rocket exhaust is at a high temperature and gives up most of its heat energy to the air with which it mixes, increasing the temperature of the latter. Further, it will be assumed that the entire effect of burning the jet gases, or additional fuel sprayed into the middle section of the duct, is to add heat to the working fluid. Hence it is seen that the processes to which the rocket gases are subjected in the combustion chamber can be identified with a certain quantity of heat added to the working fluid.

The ducted rocket thrust calculations are conveniently separated into three groups (listed below) distinguished by the amount of heat H added to the working fluid, over and above that contributed by the rocket jet in the simple mixing process.

(i) <u>No heat added</u>

For this situation it is assumed that none of the heat energy stored in the combustible components of the exhaust is released; that is, the ducted rocket is considered to be strictly a thrust augmentor in the usual sense. Mathematically, the condition that no heat (other than that present in the rocket jet) is added to the working fluid is expressed by

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setting $H=0$ in Eq (8), computing v_3'', and continuing the calculations with the corresponding values of ρ_3'', p_3'', etc. As observed above, for this case the ducted rocket is essentially a thrust augmentor; hence the results of the calculations indicate the performance of a high-speed thrust augmentor (Cf Fig 2).

(ii) Complete burning of the propellant gases

In this instance it is assumed that the burning of the rocket jet gases is fully realized; that is, all of the theoretical heat of combustion of the CO and H_2 in the rocket exhaust is considered to be released when these substances combine with the oxygen in the entering air.

The condition that the propellant gases are completely burned in the air flowing through the duct is mathematically expressed by setting $H = 10,892,000$ ft-lb, the heat of combustion of the CO and H_2 present in the rocket exhaust.

Figures 3 through 11 reveal the extent to which the thrust of a normal rocket motor can be increased when the jet is enclosed and completely burned in a ramjet duct.

The first figure (Cf Fig 3) shows the dependence of augmentation on the velocity with which the air enters the ramjet diffuser. For any given value of M_2 the augmentation clearly increases with the free flight Mach number. With a fixed free flight velocity the augmentation at first increases as the Mach number at the combustion chamber entrance increases,

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and then begins to decrease, due to a reduced pressure in the ramjet nozzle entrance at the higher flight speeds.

Figures 4 through 7 indicate the performance of a ducted rocket using a duct large enough to permit the entry of more air than that required for complete secondary combustion.

The simplest and most practical ducted rocket would employ a duct with fixed dimensions. Then, with a given diffuser entrance, an increase in the flight velocity would cause a greater mass of air to flow through the combustion chamber in a given time.

The performance results presented in Figs 4 through 7 are of value because they show conclusively that with a given rocket motor and ramjet duct the ducted rocket thrust increases with the forward velocity of the combination - or with the amount of air passing through the duct - primarily as a straight augmentation effect.

The general result that at low flight speeds the thrust of the normal rocket motor suffers a loss when enclosed in a ramjet duct (Cf Fig 2) appears also in the thrust calculations for this case, although the decrease in thrust is much smaller than it is in (i).

If it is found necessary or useful to employ a rocket in the ramjet duct - as an aid to combustion, as an instrument to facilitate the early stages of ramjet flight, or as a booster for superperformance - the small propelling force at low speeds will be detrimental.

To eliminate the lost thrust at small velocities, it may be necessary to advance to a design employing, in the initial period of the flight, a rocket so far aft that its nozzle projects behind the duct, being brought forward to the combustion chamber entrance when the flight velocity is high enough so that an actual augmentation of thrust can be realized.

(iii) Maximum possible amount of heat added

The amount of heat that can be absorbed by the working fluid is limited by the theoretically derived and experimentally observed fact that the fluid velocity at the combustion chamber exit cannot exceed the local acoustic speed (when the heat applied results from a continuous exothermic reaction). For the values of M_2 considered here, the latent heat of reaction of the Co and H_2 in the exhaust is always less than that required to accelerate the working fluid to the local velocity of sound at the combustion chamber exit. The additional heat needed can be obtained by burning gasoline, kerosene, or a similar fuel, injected into the combustion chamber.

The peak value of H is obtained by setting the radical in Formula (8) equal to zero, giving

$$H_{MAX.} = \frac{(\rho_2 f_3'' + m_2 v_2 + m_2' v_2')^2}{2(m_2 + m_2')\frac{\gamma''^2 - 1}{\gamma''^2}} - (m_2 E_2 + m_2' E_2').$$

Since the value of $H_{MAX.}$ is not employed in the later calculations, while the size of v_3'' is a fundamental intermediate result, it is more useful to compute $v_{3_{MAX.}}''$.

Again from Formula (8),

$$V_{3_{MAX.}}'' = \frac{\gamma''(\rho_2 f_3'' + m_2 v_2 + m_2' v_2')}{(m_2 + m_2')(1+\gamma'')} .$$

Mathematically, the condition that the maximum possible quantity of heat is added to the working fluid is imposed by setting

$$H_{MAX.} = \frac{(\rho_2 f_3'' + m_2 v_2 + m_2' v_2')^2}{2(m_2 + m_2')(\gamma''^2 - 1)/\gamma''^2} - (m_2 E_2 + m_2' E_2') ,$$

since this value of H makes the radical in (8) zero, causing V_3'' to assume its maximum size.

Figures 2 and 3 have as their subjects ducted rockets with variable diffuser and combustion chamber areas, so that only a preassigned amount of air is allowed to enter the duct (for example, only so much air as is required to support the complete combustion of the rocket exhaust gases at some given flight velocity), and a variable nozzle designed to expel the products of the secondary reaction at atmospheric pressure under all flight conditions.

However, in Figs 8 through 12 the dimensions of the ramjet diffuser and combustion chamber are supposed constant, the nozzle opening alone being considered variable. Figures 8 through 11 show the results of a numerical example in which the diffuser area ratio f_2/f_0 is allowed to assume the values 1.5, 2, 3, and 4, while the combustion chamber area is fixed at 4 sq ft.

The extent to which the amount of heat added to the working fluid affects the augmentation of thrust of the normal rocket motor is revealed in Figs 8, 9, 10, and 11. Each graph covers a range of free flight Mach numbers, $0.12 \leq M_0 \leq 1$,* for three cases:

*For a combustion chamber area of 4 sq ft, the value of M_0 at which just enough air to support complete secondary burning of the exhaust gases enters is given by $0.0245\, f_2/f_0$. At the lower limit of M_0, 0.12, sufficient air for complete combustion enters the duct when $f_2/f_0 < 5$.

(a) No heat is added, other than that present in the rocket jet when it begins to mix with the air in the duct; the performance of the ducted rocket in this case is represented by the curve labeled "NO BURNING (H=0)".

(b) All of the energy stored in the jet gases is liberated (by burning the CO and H_2 components of the rocket exhaust) and added to the air flowing through the duct; the performance of the ducted rocket in this case is depicted by the curve labeled "COMPLETE SECONDARY BURNING OF THE ROCKET EXHAUST (H = 10,892,000 FT-LB)".

(c) A quantity of heat sufficient to bring the velocity of the working fluid up to that of sound is added to the mixture, this amount of heat being the maximum it is possible to transfer to the fluid; the performance curve in this case is labeled "SUFFICIENT HEAT TO MAKE $M_3'' = 1$".

Inspection of the figures under discussion reveals that the ducted rocket thrust may be considerably multiplied if fuel such as gasoline is injected into the combustion chamber at high flight speeds, the additional augmentation of thrust depending on the area ratio f_2/f_0. The curves labeled "SUFFICIENT HEAT, etc.," in Figs 8 through 11 reveal the extent to which such increase in thrust follows from the addition of the maximum possible amount of heat to the air flowing through the ramjet duct. However, Figs 8 through 11 show clearly that large values of H are not always to be desired, since the duct-jet thrust when the rocket exhaust gases are completely burned (H = 10,892,000 ft-lb) is considerably greater than that obtained with a sonic velocity at the end of the combustion chamber, even though the heat required in the latter case is invariably greater than 10,892,000 ft-lb.

Figure 12 offers a comparison between the performance of a ducted rocket and a ramjet of equal physical dimensions consuming fuel at the same rate as the rocket motor.

The performance of such an "equivalent" ramjet may be calculated in the elementary manner of the ducted rocket by recognizing that in the two propulsion systems the only essential distinction is the momentum of the entering fuel, the exhaust gases having a momentum far in excess of that with which fresh fuel is injected into the ramjet. Further, the rocket of the ducted rocket unit required m_2' slugs of propellants per sec (fuel and oxidant), while the ramjet needs only the fuel component (of mass equal to m_{2r}').

That is, the three conditions of conservation of mass, conservation of pressure forces and momentum, and conservation of energy [Cf Eqs (1), (2), and (3), pp. 23 and 24] , applied to the "equivalent" ramjet, become

$$m_2 + m_2' = \rho_3'' f_3'' v_3'' \qquad (1_r)$$

$$m_2 v_2 = (m_2 + m_2') v_3'' + (p_3'' - p_2) f_3'' \qquad (2_r)$$

$$m_2 E_2 + m_{2r}' E_{2r}' + H = \rho_3'' f_3'' v_3'' \left(\frac{\gamma''}{\gamma''-1} \frac{p_3''}{\rho_3''} + \frac{1}{2} v_3''^2 \right) \qquad (3_r)$$

where

$$E_2 = \frac{\gamma}{\gamma-1} \frac{p_2}{\rho_2} + \frac{1}{2} v_2^2 , \qquad E_{2r}' = \frac{\gamma'}{\gamma'-1} \frac{p_2}{\rho_2'} \quad *$$

*Actually, E_{2r}' should be given as $\gamma_r' p_2 / (\gamma_r' - 1) \rho_{2r}'$. However, the error committed in assuming that $\rho_{2r}' = \rho_2'$ will not influence the results to an extent greater than the other approximations used in the analysis. This follows from the fact that the density of the rocket exhaust jet at the beginning of mixing, ρ_2' , will be small because of the high temperature of the gas; since any new fuel injected into the combustion chamber enters a body of air at a high temperature, the density ρ_{2r}' will also be small. The values of γ' for the rocket exhaust jet and the vaporized new fuel are also assumed to be the same.

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and

m'_{2r} = mass of the fuel component of the rocket propellants.

The solution of the simultaneous equations (1_r), (2_r), (3_r) for the unknowns V''_3, ρ''_3, and P''_3 gives three formulae similar to those derived in the ducted rocket calculations:

$$V''_3 = \frac{\frac{\gamma''}{1+\gamma''}}{m_2+m'_{2r}}\left[P_2 f''_3 + m_2 V_2 - \sqrt{(P_2 f''_3 + m_2 V_2)^2 - 2\frac{\gamma''^2-1}{\gamma''^2}(m_2+m'_{2r})(m_2 \epsilon_2 + m'_{2r}\epsilon'_{2r} + H)}\right] \qquad (8_r)$$

$$\rho''_3 = \frac{m_2 + m'_{2r}}{f''_3 V''_3} \qquad (9_r)$$

$$P''_3 = \frac{P_2 f''_3 + m_2 V_2}{f''_3} - \frac{m_2 V_2 + m'_{2r}}{f''_3} V''_3 . \qquad (10_r)$$

When the rocket propellant is specialized to the propellant combination of the previous calculations, utilizing liquid oxygen-gasoline in the ratio 3:2 by weight, $m'_{2r} = 0.4\, m'_2$. In the ducted rocket study a 500 lb thrust rocket motor, for which $m'_2 = 0.0777$ slugs per sec, was consistently used, so the "equivalent" ramjet will have a fuel consumption of 0.0311 slugs of gasoline per sec.

Since the fuel mass flow of the equivalent ramjet is different from the propellant mass flow of the basic rocket motor, the augmentation factor is in this case the ratio

$$\frac{\text{effective exhaust velocity}}{16{,}100 \text{ ft/sec}}.$$

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Once the physical properties of the ramjet working fluid at the nozzle entrance are known, the remaining calculations, leading to the computation of thrust and augmentation factor, are identical with those previously detailed.

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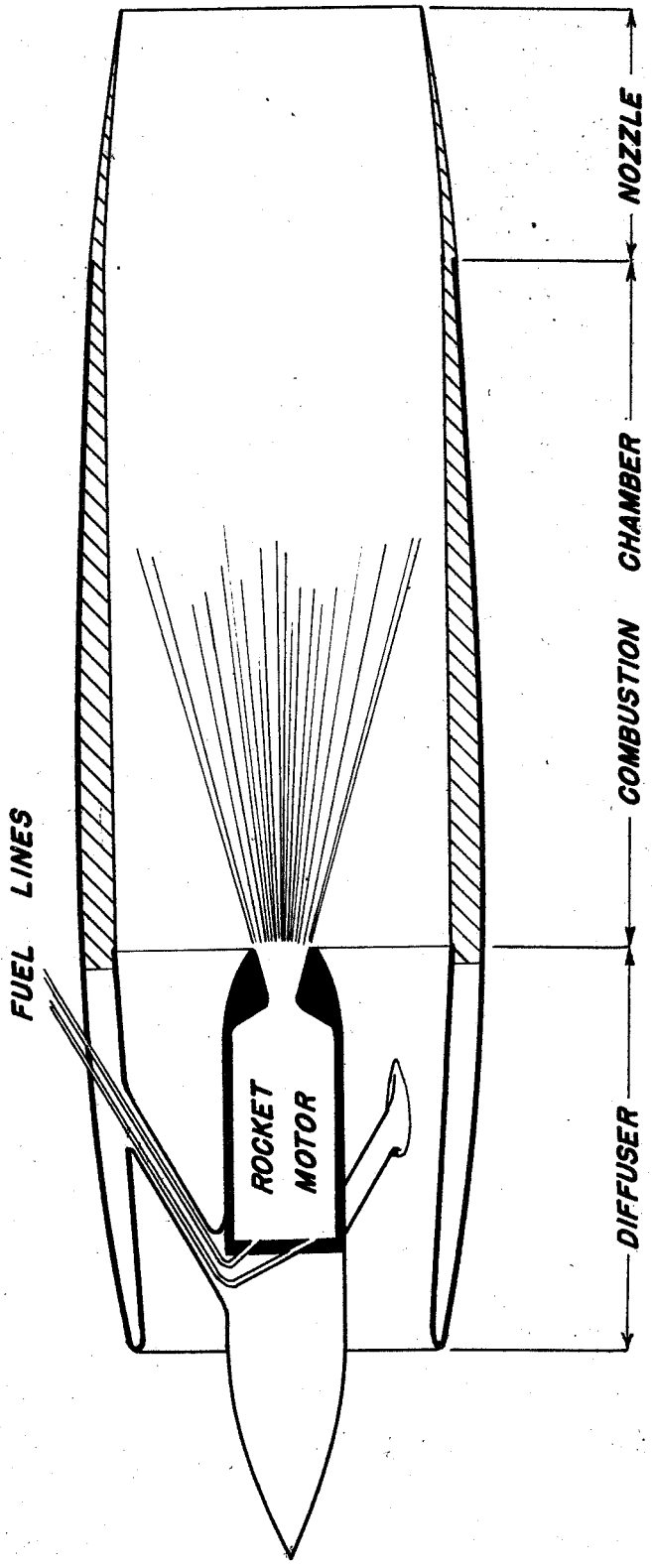

FIGURE 1 - CROSS SECTION THROUGH DUCTED ROCKET

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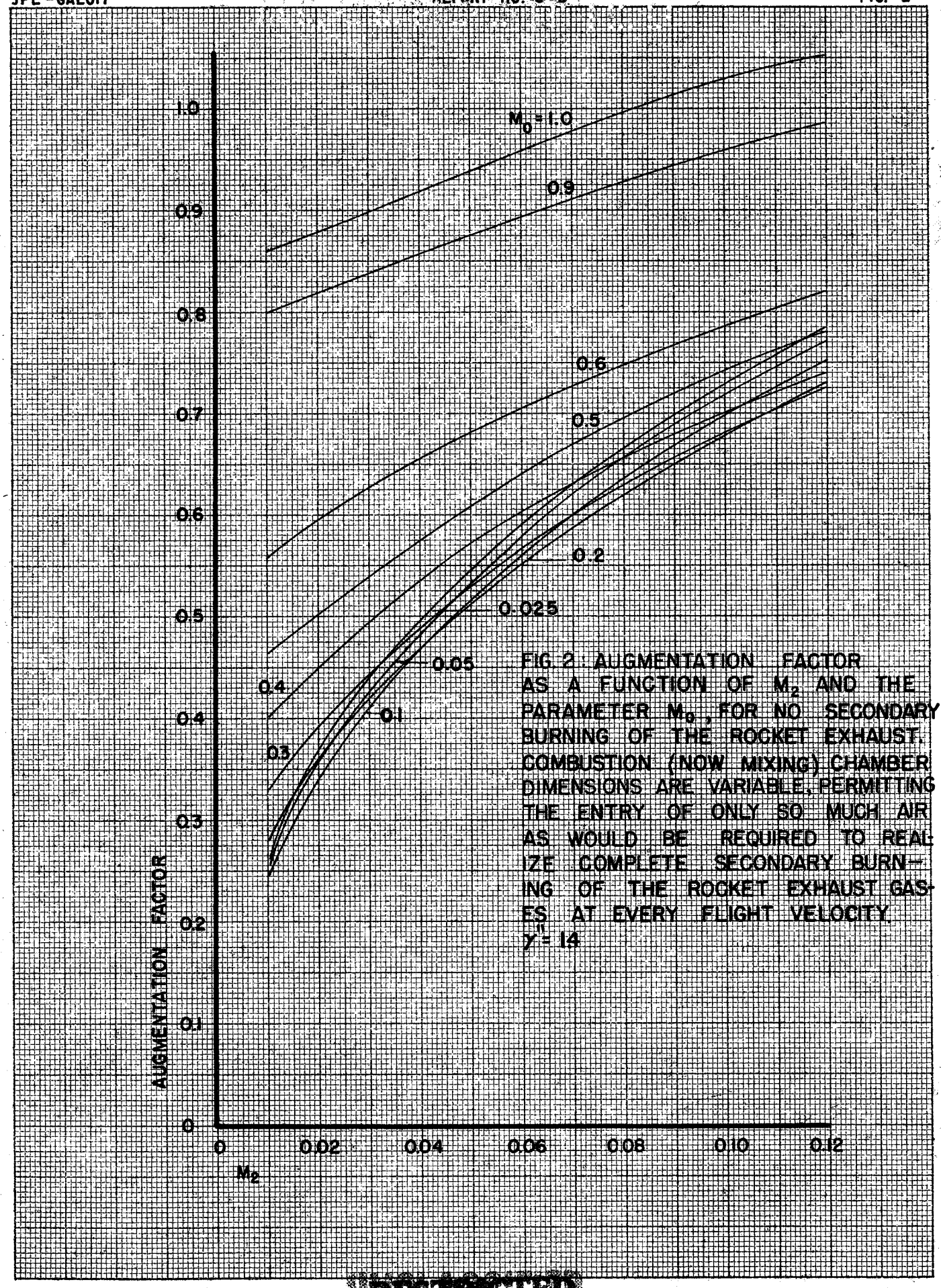

FIG. 2: AUGMENTATION FACTOR AS A FUNCTION OF M_2 AND THE PARAMETER M_0, FOR NO SECONDARY BURNING OF THE ROCKET EXHAUST. COMBUSTION (NOW MIXING) CHAMBER DIMENSIONS ARE VARIABLE, PERMITTING THE ENTRY OF ONLY SO MUCH AIR AS WOULD BE REQUIRED TO REALIZE COMPLETE SECONDARY BURNING OF THE ROCKET EXHAUST GASES AT EVERY FLIGHT VELOCITY. $\gamma'' = 1.4$

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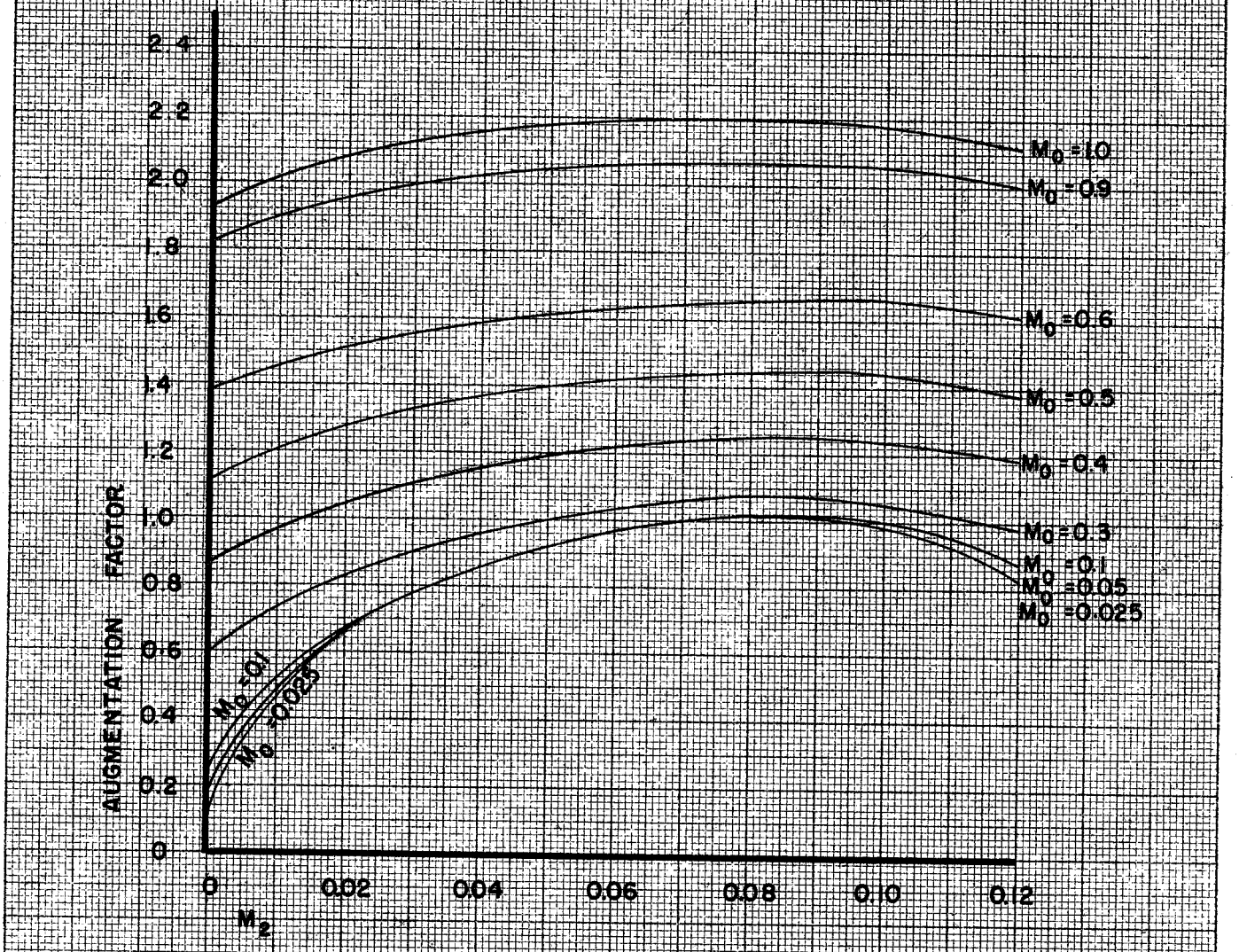

FIG. 3: AUGMENTATION FACTOR AS A FUNCTION OF M_2 AND THE PARAMETER M_0. COMBUSTION CHAMBER DIMENSIONS ARE VARIABLE, PERMITTING THE ENTRY OF JUST ENOUGH AIR TO REALIZE COMPLETE SECONDARY BURNING OF THE ROCKET EXHAUST GASES AT EVERY FLIGHT VELOSITY. $\gamma'' = 1.4$

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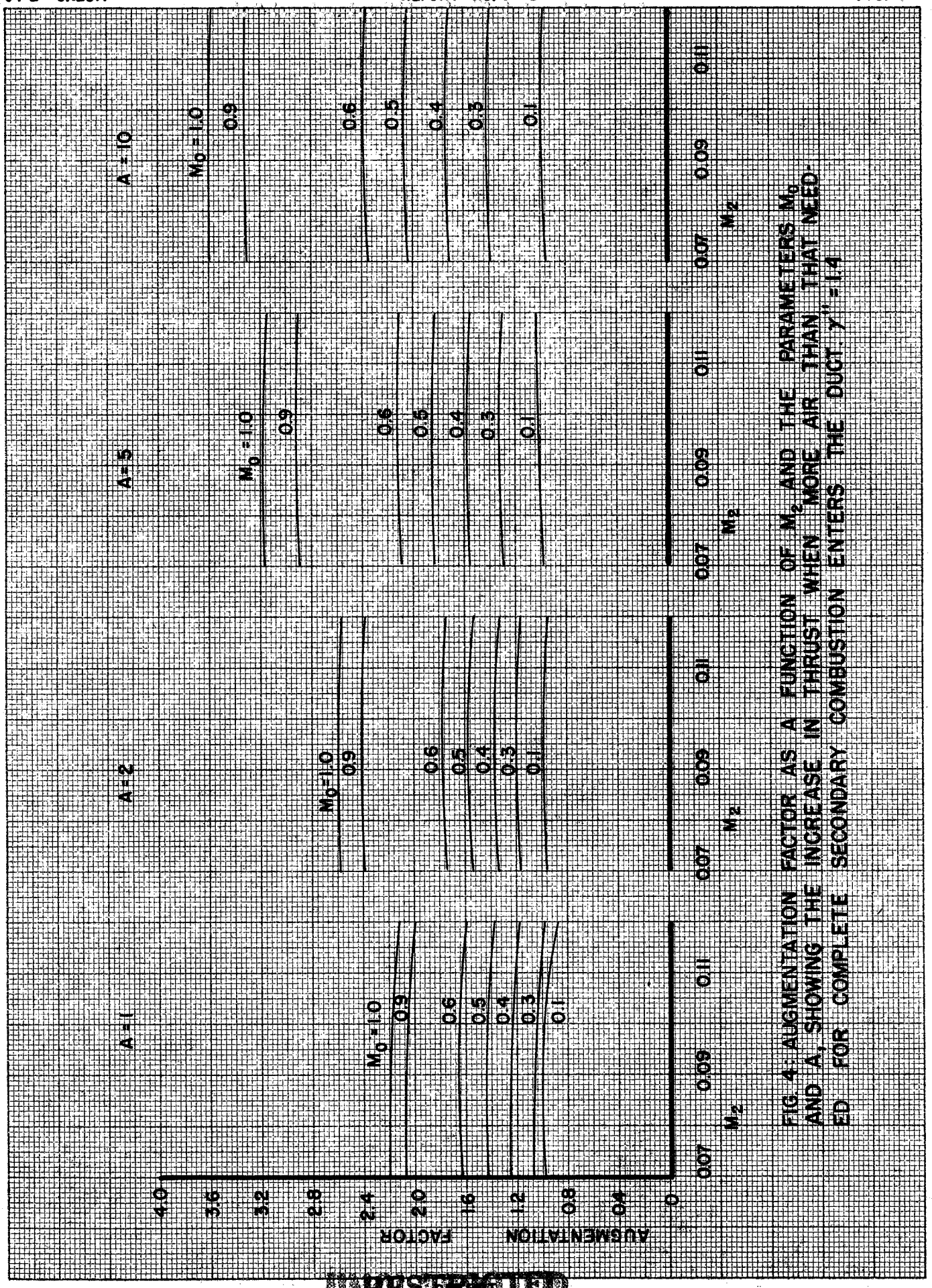

FIG. 4: AUGMENTATION FACTOR AS A FUNCTION OF M_2 AND THE PARAMETERS M_0 AND A, SHOWING THE INCREASE IN THRUST WHEN MORE AIR THAN THAT NEED-ED FOR COMPLETE SECONDARY COMBUSTION ENTERS THE DUCT. $\gamma'' = 1.4$

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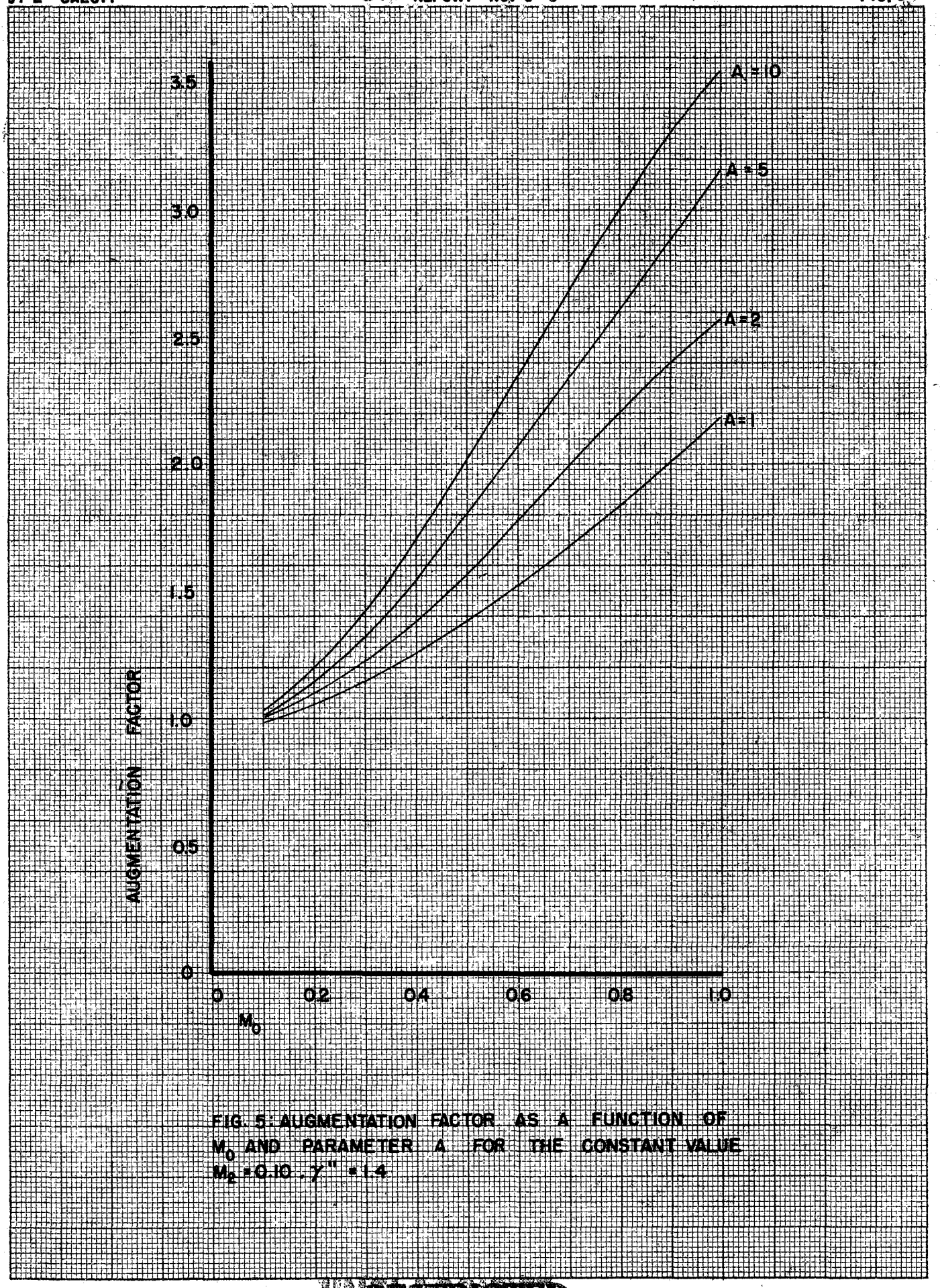

FIG. 5: AUGMENTATION FACTOR AS A FUNCTION OF M_0 AND PARAMETER A FOR THE CONSTANT VALUE $M_2 = 0.10$, $\gamma'' = 1.4$

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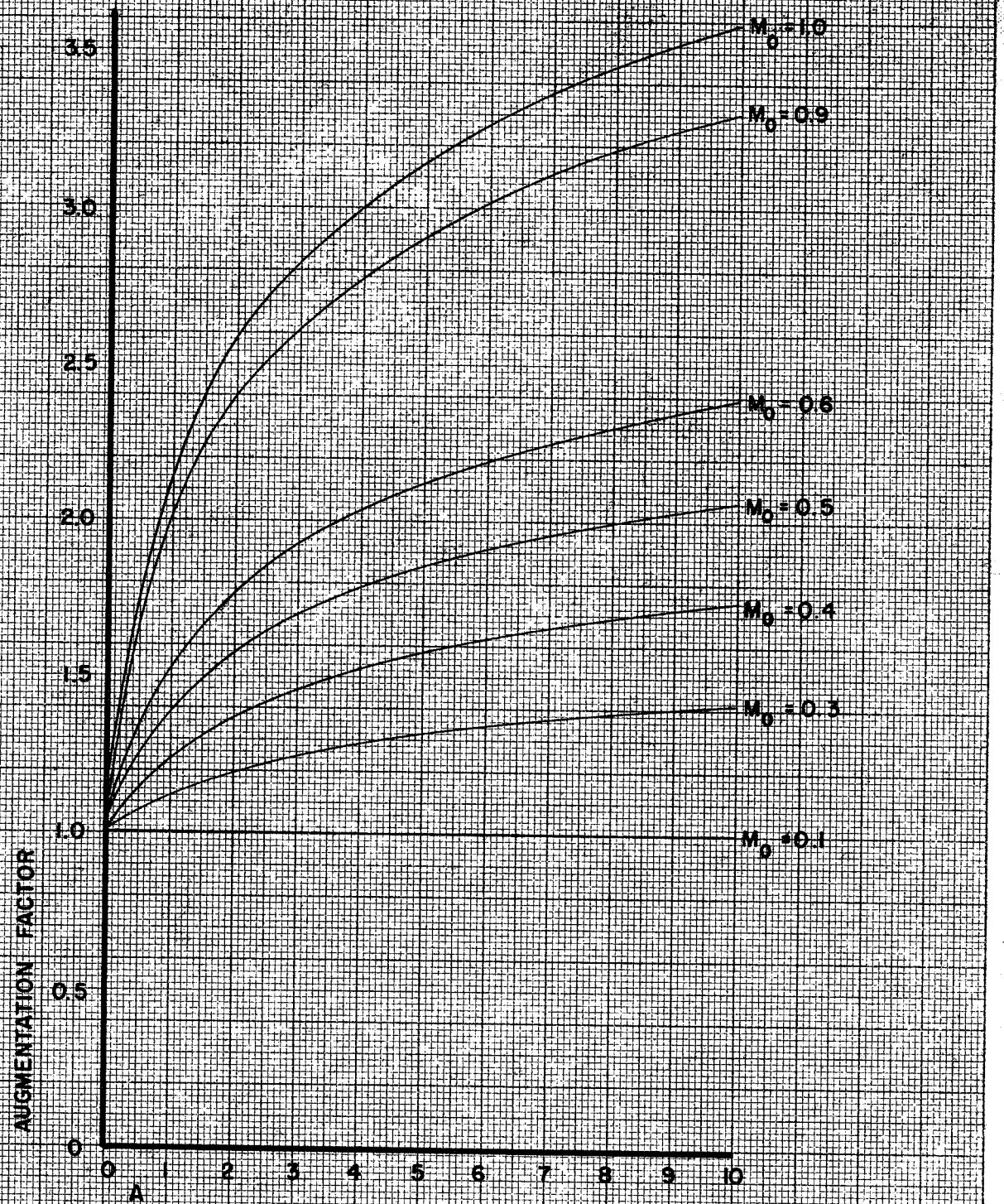

FIG. 6: AUGMENTATION FACTOR AS A FUNCTION OF A AND THE PARAMETER M_0 FOR THE CONSTANT $M_2 = 0.10$, $\gamma'' = 1.4$

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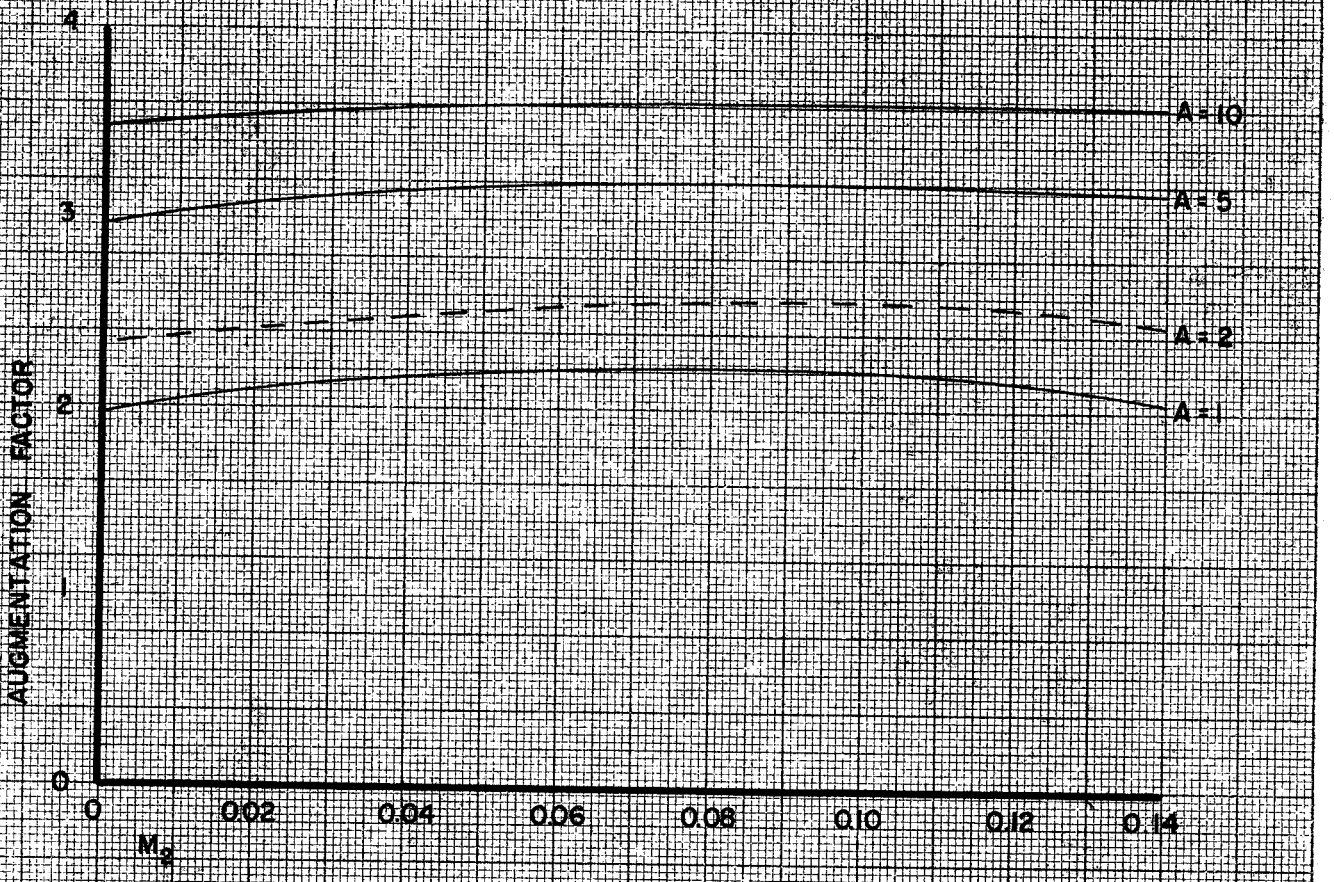

FIG. 7: AUGMENTATION FACTOR AS A FUNCTION OF M_2 AND THE PARAMETER A FOR THE CONSTANT VALUE $M_0 = 10, \gamma'' = 1.4$

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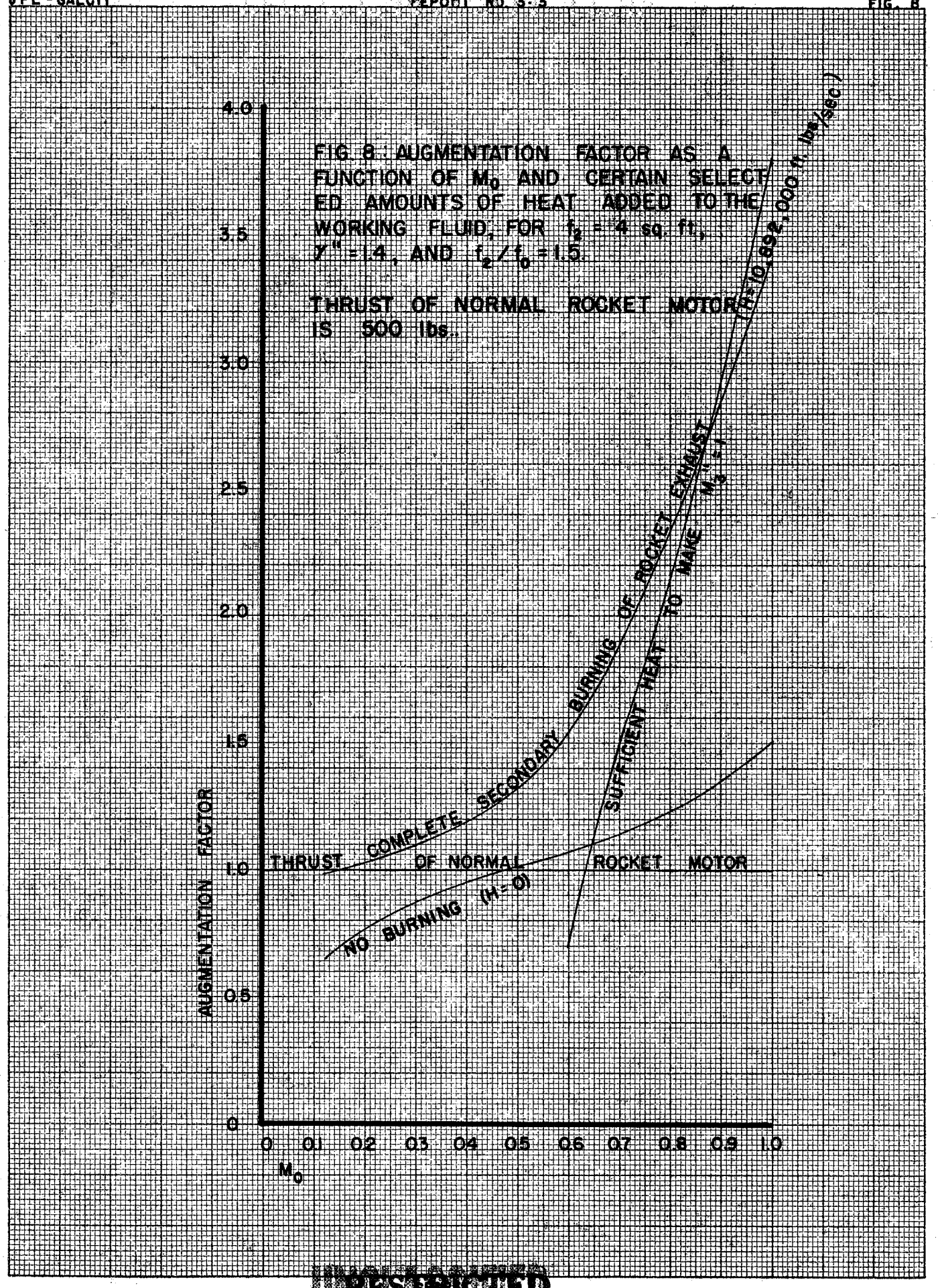
FIG. 8. AUGMENTATION FACTOR AS A FUNCTION OF M_0 AND CERTAIN SELECTED AMOUNTS OF HEAT ADDED TO THE WORKING FLUID, FOR $f_2 = 4$ sq. ft., $\gamma'' = 1.4$, AND $f_2/f_0 = 1.5$.

THRUST OF NORMAL ROCKET MOTOR IS 500 lbs.



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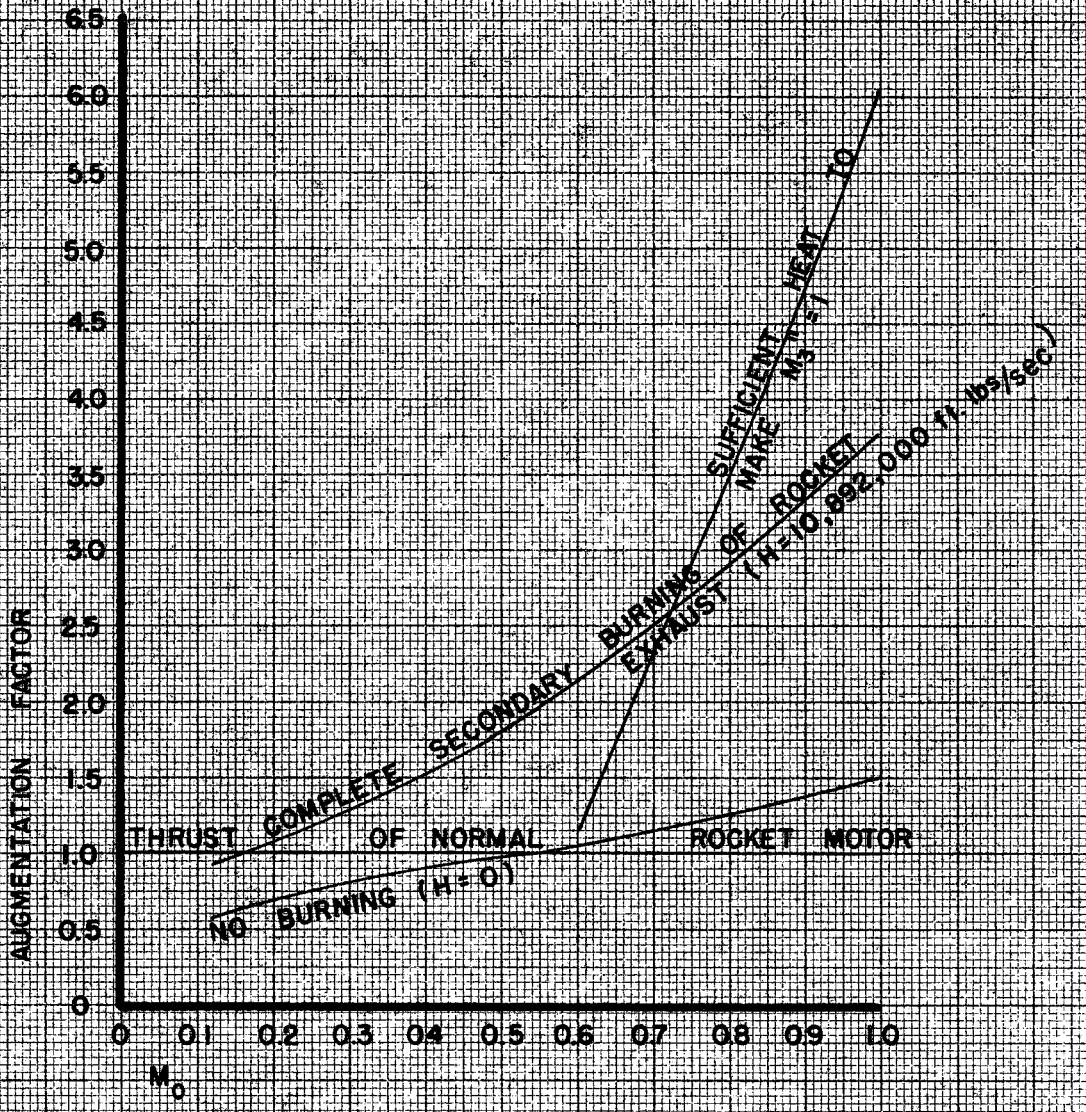

FIG. 9: AUGMENTATION FACTOR AS A FUNCTION OF M_0 AND CERTAIN SELECTED AMOUNTS OF HEAT ADDED TO THE WORKING FLUID, FOR $f_2 = 2$, $\gamma'' = 1.4$, AND $f_2/f_0 = 2$

THRUST OF NORMAL ROCKET MOTOR IS 500 lbs.

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FIG. 10: AUGMENTATION FACTOR AS A FUNCTION OF M_0 AND CERTAIN SELECTED AMOUNTS OF HEAT ADDED TO THE WORKING FLUID, FOR $f_2 = 4$ sq. ft., $\gamma'' = 1.4$, AND $f_2/f_0 = 3$.

THRUST OF NORMAL ROCKET MOTOR IS 500 lbs.

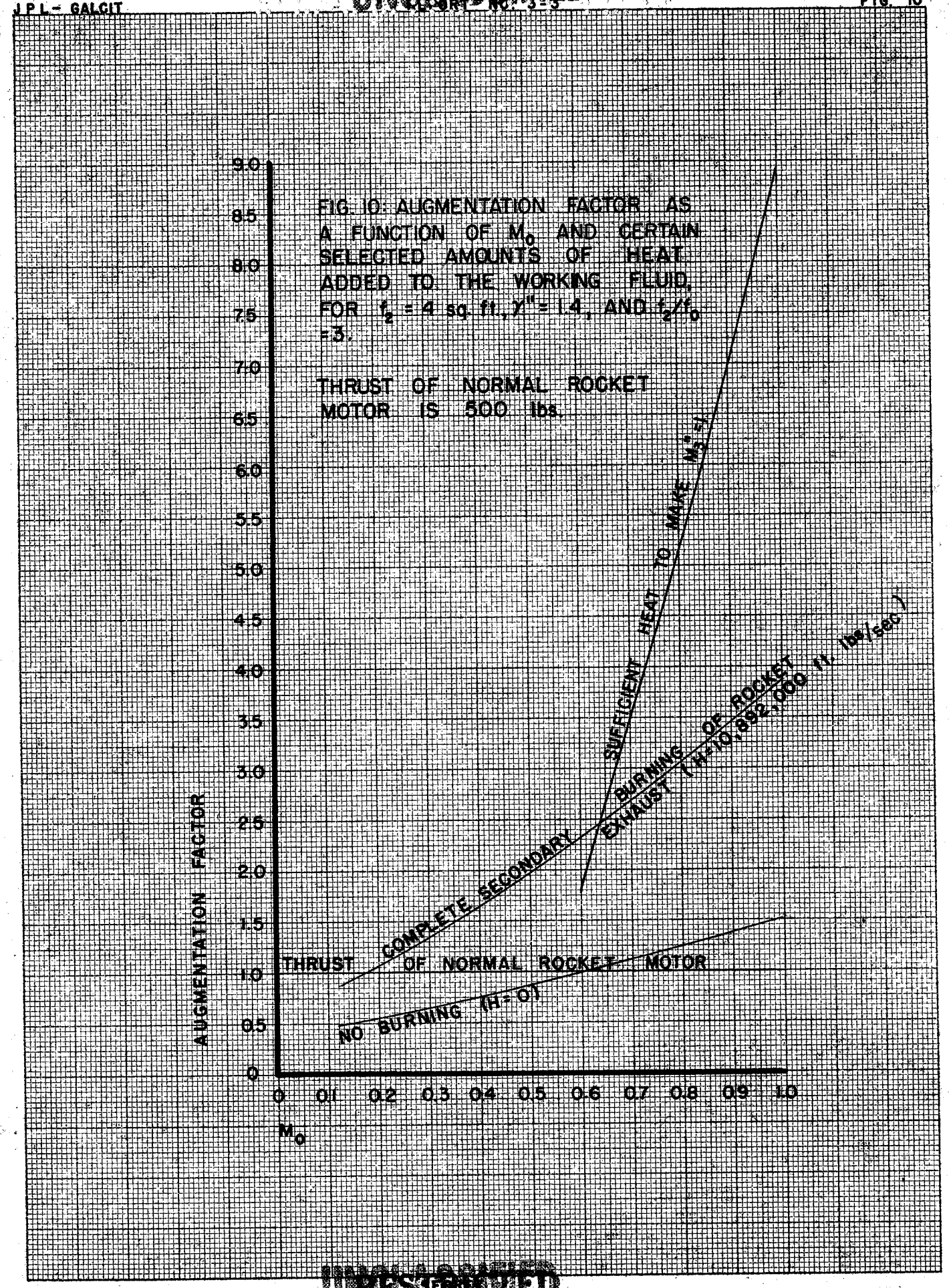

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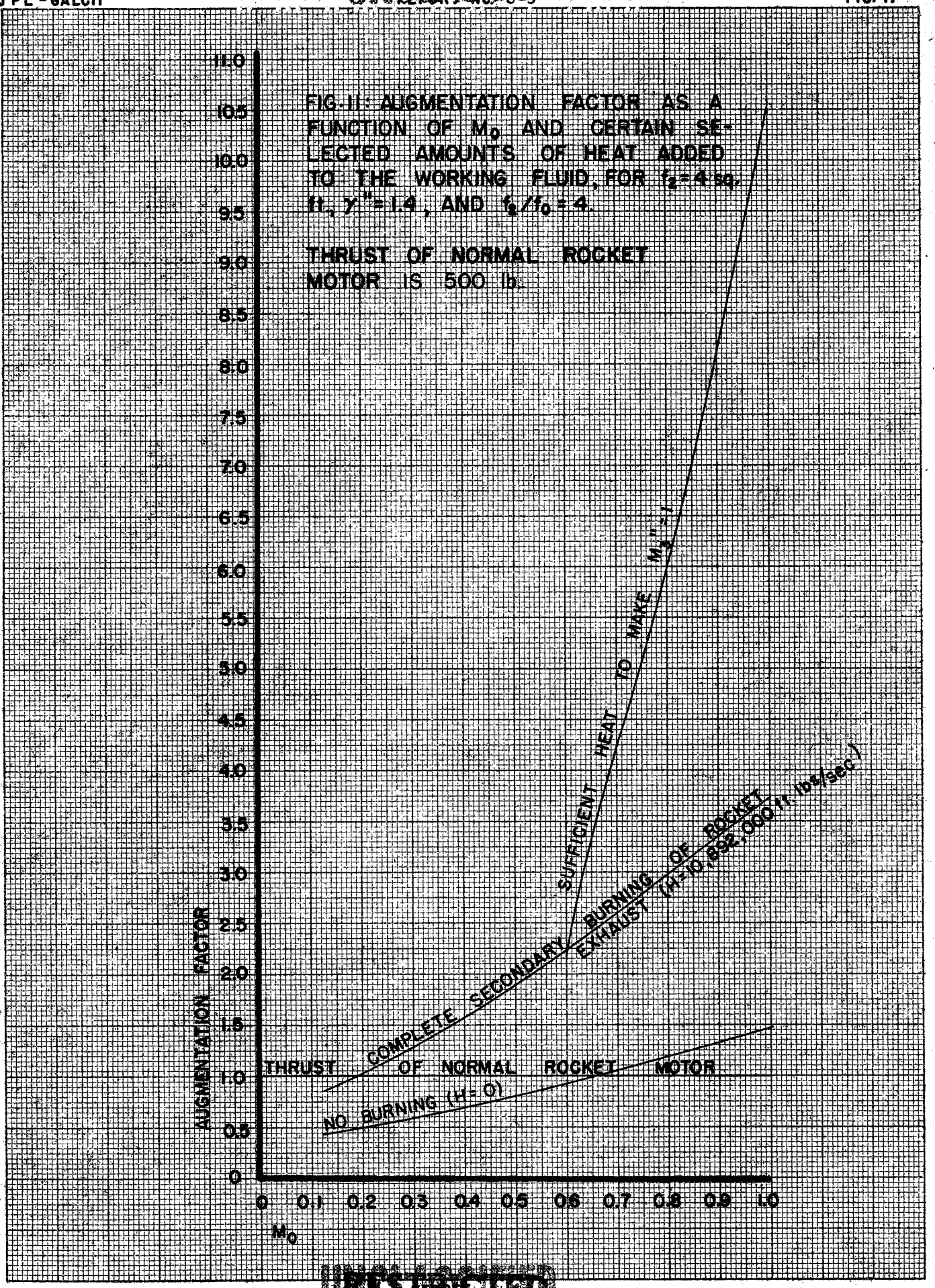
FIG. 11: AUGMENTATION FACTOR AS A FUNCTION OF M_0 AND CERTAIN SELECTED AMOUNTS OF HEAT ADDED TO THE WORKING FLUID, FOR $f_2 = 4$ sq. ft., $\gamma'' = 1.4$, AND $f_2/f_0 = 4$.

THRUST OF NORMAL ROCKET MOTOR IS 500 lb.



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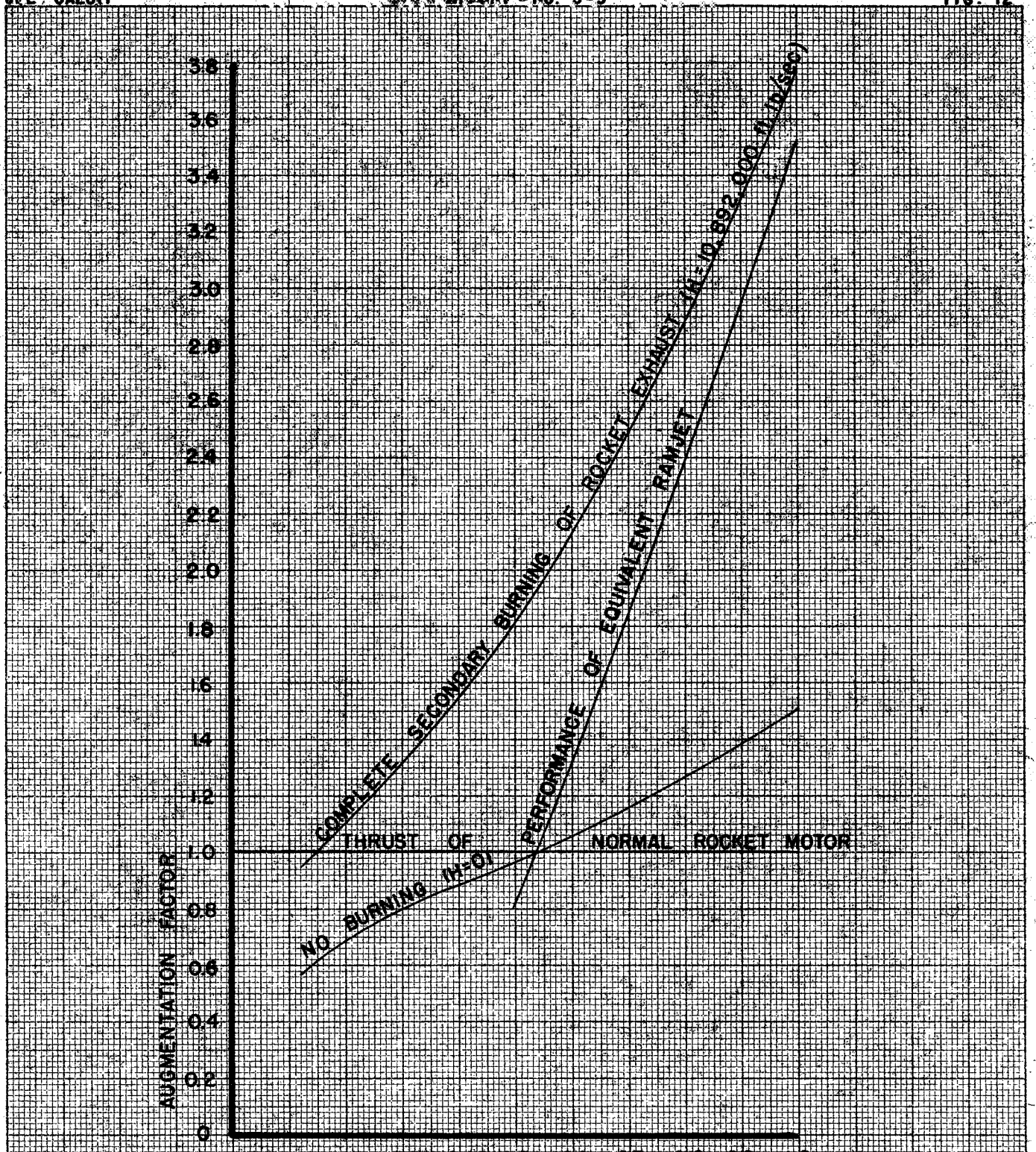

FIG. 12 COMPARISON OF AUGMENTATION FACTORS, DUCTED ROCKET AND RAMJET OF EQUAL PHYSICAL DIMENSIONS AND FUEL CONSUMPTION, WHEN $f_2 = 4$ sq. ft., $\gamma'' = 1.4$, AND $f_2/f_0 = 2$.

THRUST OF NORMAL ROCKET IS 500 lb.

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